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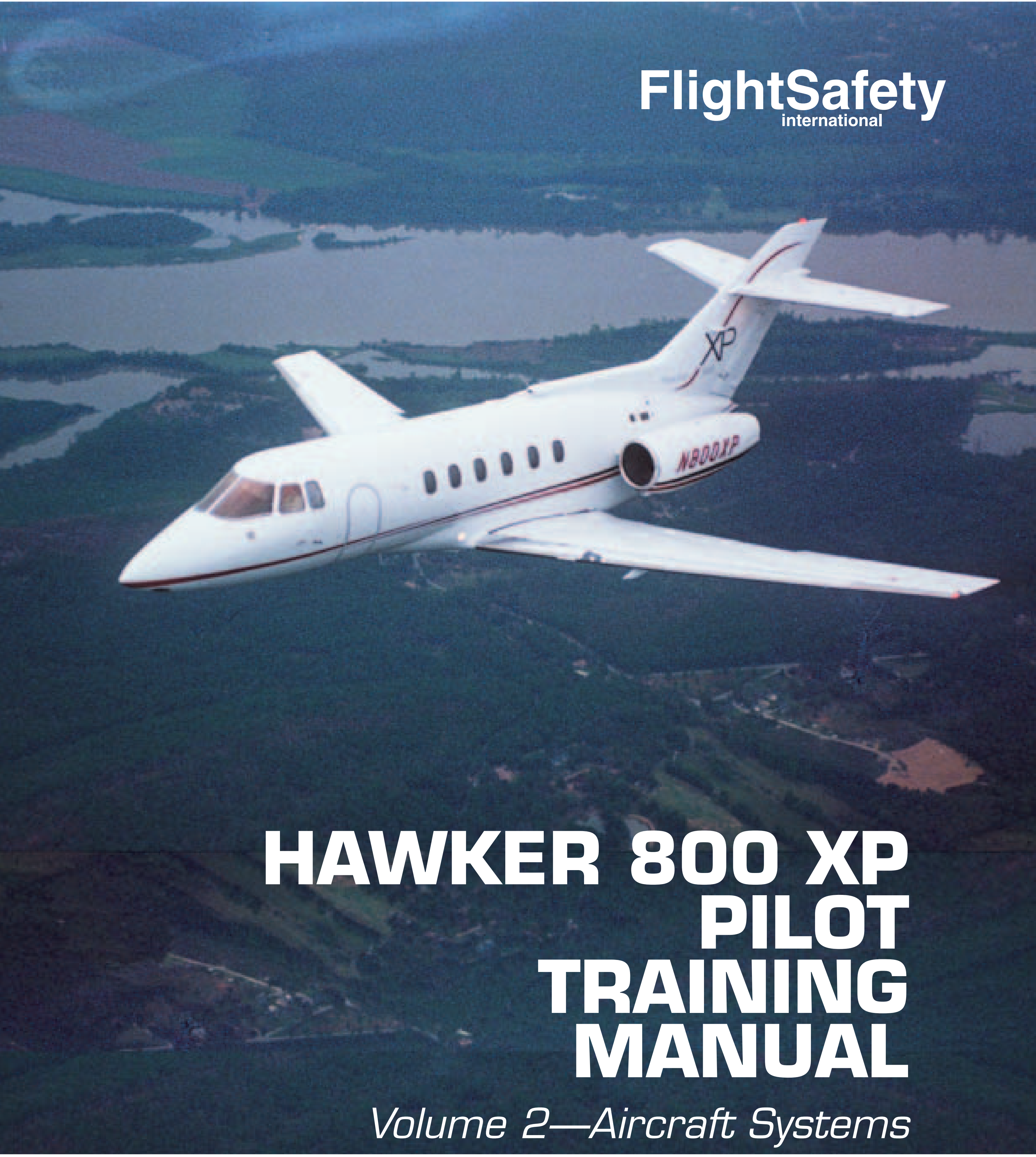
HAWKER 800 XP PILOT TRAINING MANUAL VOLUME 2

Record of Revision No. 1.02

This is a complete reprint of the *Hawker 800 XP Pilot Training Manual.*

The portion of the text or figure affected by the current revision is indicated by a solid vertical line in the margin. A vertical line adjacent to blank space means that material has been deleted. In addition, each revised page is marked "Revision 1.02" in the lower left or right corner.

The changes made in this revision will be further explained at the appropriate time in the training course.



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HAWKER 800 XP PILOT TRAINING MANUAL

Volume 2—Aircraft Systems

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Courses for the Hawker 800 XP aircraft are taught at:

FlightSafety International
Greater Philadelphia/Wilmington Learning Center
New Castle County Airport
155 N. DuPont Highway
New Castle, DE 19720
(302) 328-7548

FlightSafety International
Houston Learning Center
William P. Hobby Airport
7525 Fauna Street
Houston, TX 77061
(713) 644-1521

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NOTICE

The material contained in this training manual is based on information obtained from the aircraft manufacturer's Pilot Manuals and Maintenance Manuals. It is to be used for familiarization and training purposes only.

At the time of printing it contained then-current information. In the event of conflict between data provided herein and that in publications issued by the manufacturer or the FAA, that of the manufacturer or the FAA shall take precedence.

We at FlightSafety want you to have the best training possible. We welcome any suggestions you might have for improving this manual or any other aspect of our training program.

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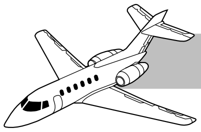


SYLLABUS COURSE INFORMATION

LEARNING CENTER INFORMATION

FlightSafety International is an aviation training company that provides type-specific training programs for over 50 different models of aircraft, using a fleet of over 150 simulators. FlightSafety operates over 38 Learning Centers, including Centers in Europe and Canada.

Training for the HS 125/800 XP is conducted at two FlightSafety Learning Centers: Greater Philadelphia/Wilmington and Houston, Texas.

Listed below are the addresses of these Centers:

Greater Philadelphia/Wilmington Learning Center
New Castle County Airport
155 North Dupont Highway
New Castle, Delaware 19720
(302) 328-7548
(800) 733-7548
Fax (302) 322-6664

Houston Learning Center
William P. Hobby Airport
7525 Fauna at Airport Boulevard
Houston, Texas 77061
(713) 644-1521
(800) 927-1521
Fax (713) 644-2118

DESCRIPTION OF TRAINING FACILITY

Each classroom and briefing room is adequately heated, lighted, and ventilated to conform to local building, sanitation, and health codes. The building construction prevents any distractions from instruction conducted in other rooms or by flight operations and maintenance operations on the airport.

Classrooms are equipped for computer graphics, 35mm slides and video presentations, controlled from a specially designed lectern. A standard overhead projector is mounted on the lectern. A white liquid chalk board is wall-mounted next to a combination projection screen/white liquid chalk board. Cockpit panel posters are on display. Briefing rooms are equipped with cockpit panel posters, a white liquid chalkboard, a table, and chairs for individual or small-group briefings. A floor plan of each Center follows.

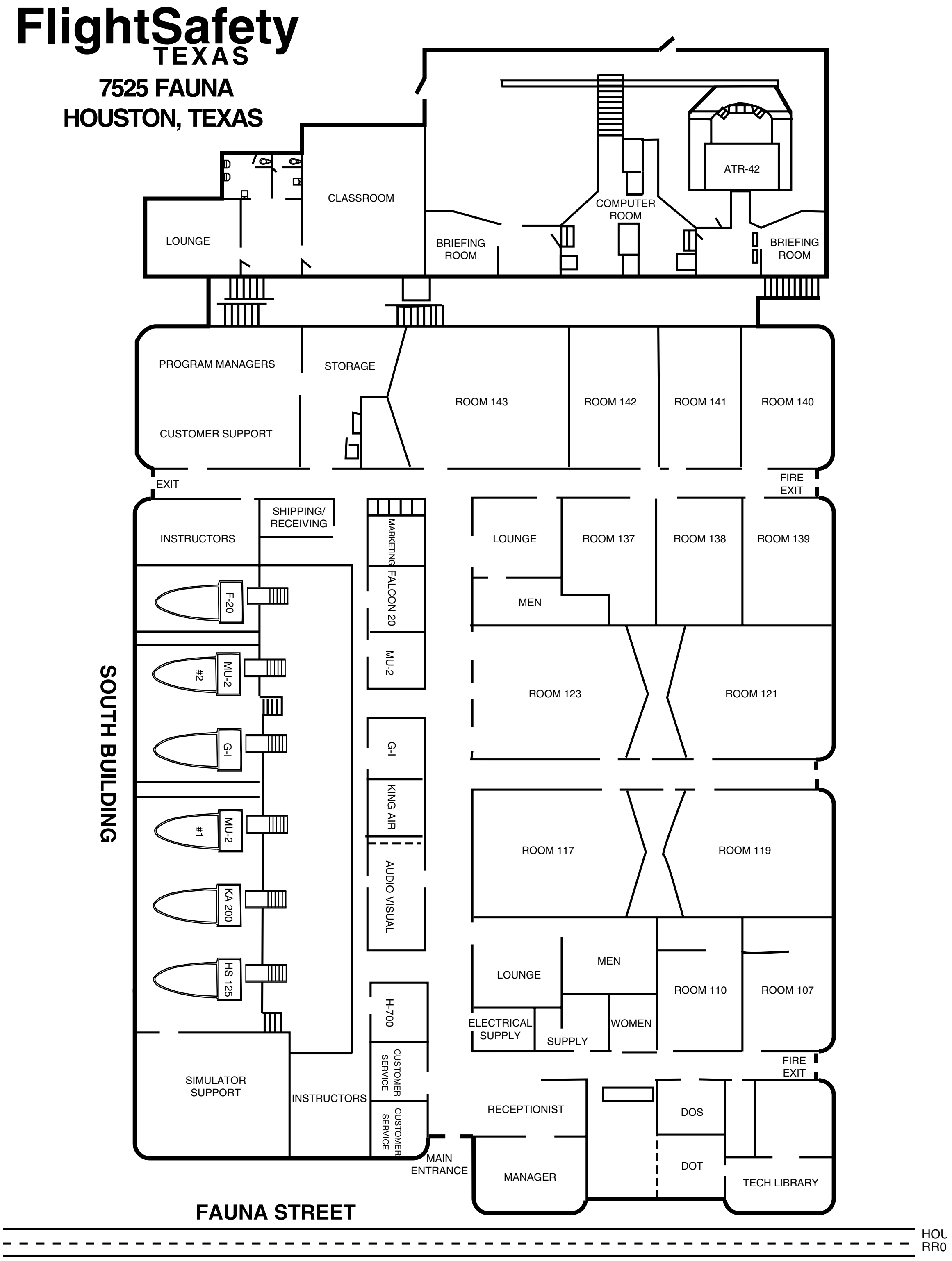


Figure SLY-1. Houston Facility Floor Plan (South Building)

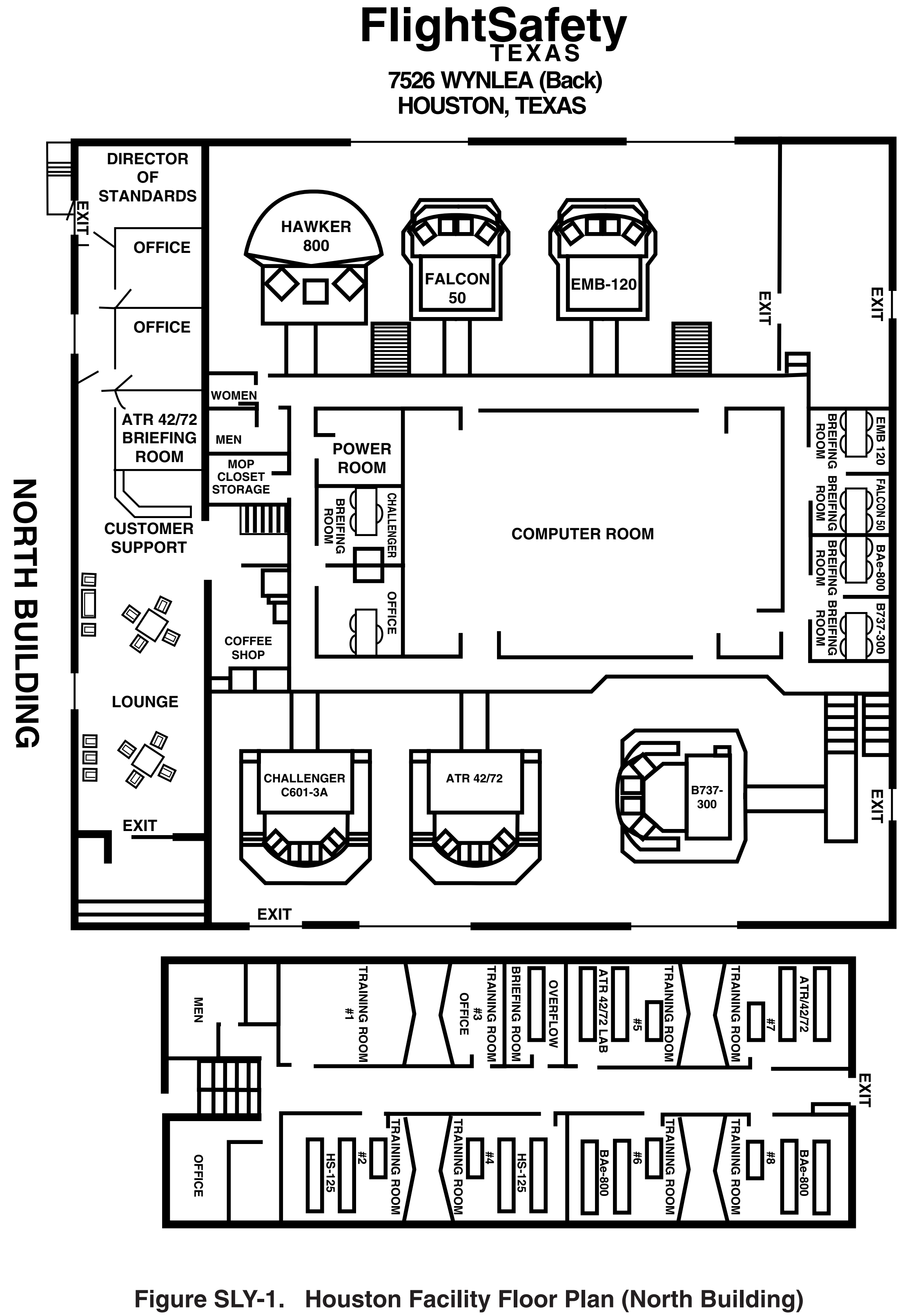
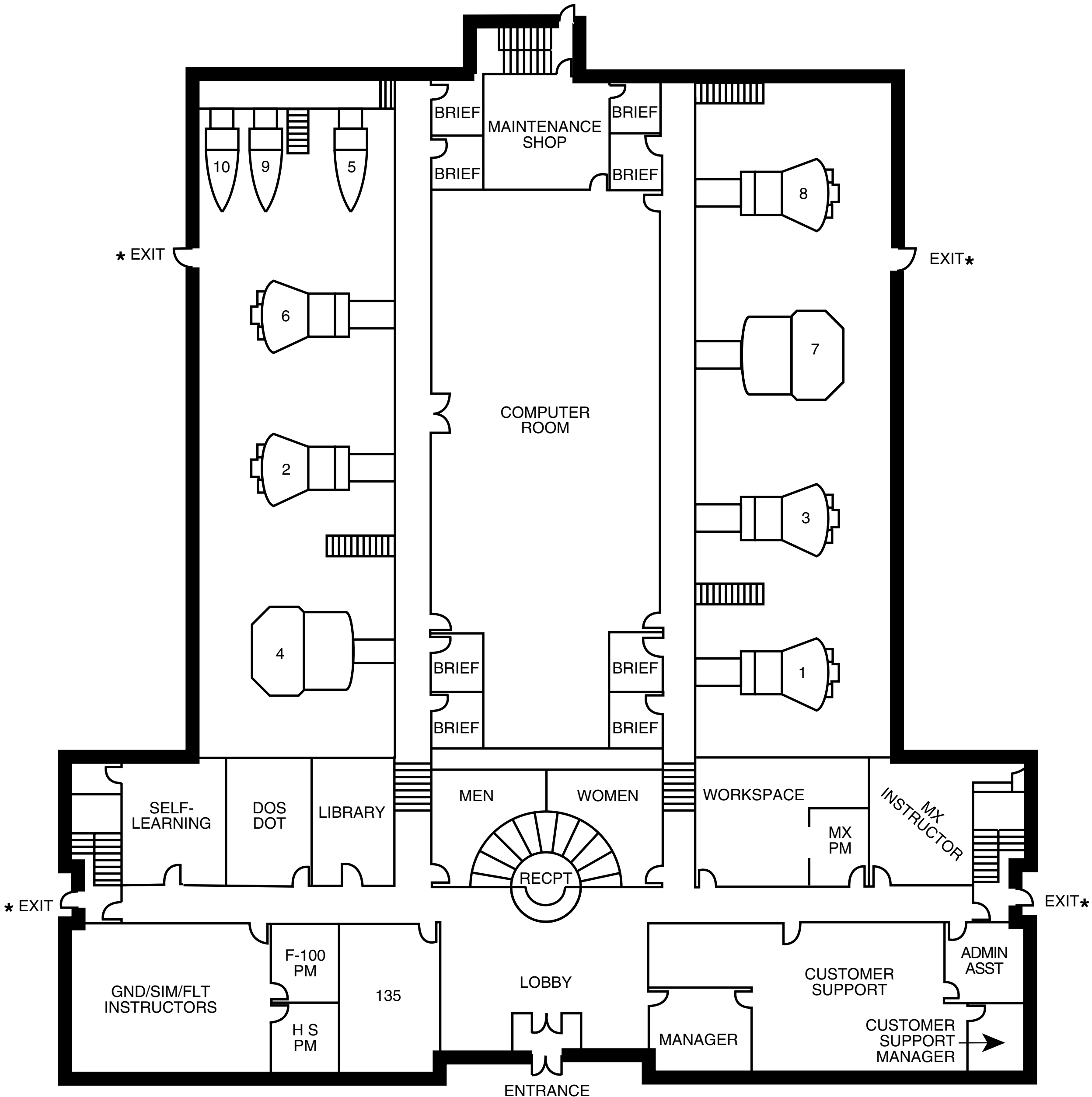


Figure SLY-1. Houston Facility Floor Plan (North Building)



SIMULATORS

1. WESTWIND II 1124
2. HAWKER (SPERRY-EQUIPPED), LEVEL C
3. HS-125/700 WITH TRs, LEVEL C
4. FUTURE
5. WESTWIND 1124 VISUAL
6. ASTRA 1125, LEVEL C
7. FOKKER 100
8. ATR 42/72
9. JET COMMANDER 1121 CPT
10. HS-125/1A-400 CPT

* FIRE EXIT

ADMINISTRATION SIMULATORS

FIRST FLOOR

Figure SLY-2. Greater Philadelphia/Wilmington Facility Floor Plan (First Floor)

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**Greater Philadelphia/
Wilmington Learning Center
Wilmington, Delaware**

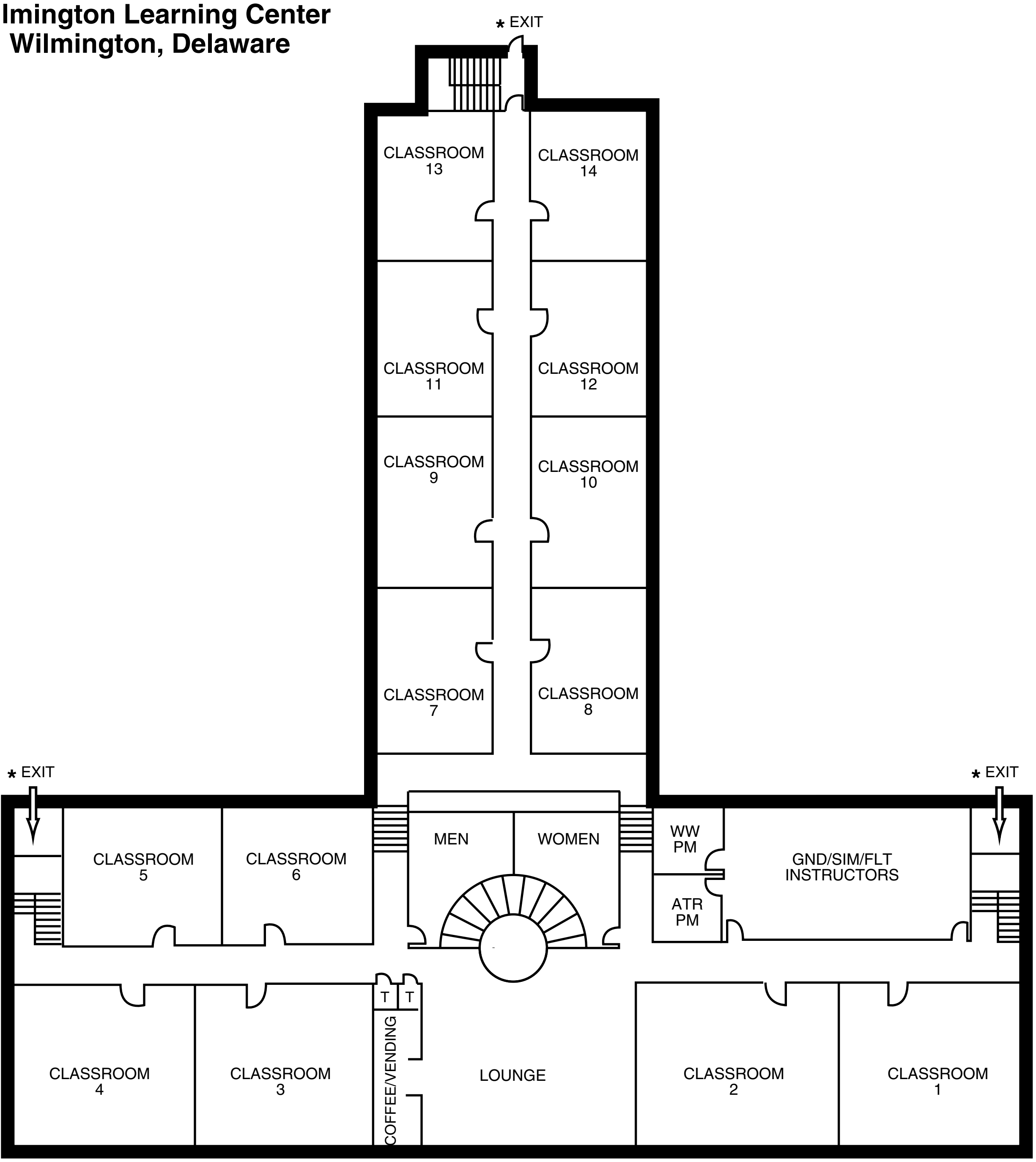


Figure SLY-2. Greater Philadelphia/Wilmington Facility Floor Plan (Second Floor)

GREATER PHILADELPHIA WILMINGTON LEARNING CENTER

Classroom Number	Size in Feet	Student Capacity
1	24 X 30	28
2	24 X 30	20
3	24 X 26	21
4	24 X 26	21
5	19 X 24	12
6	19 X 24	12
7	19 X 24	18
8	19 X 24	18
9	19 X 24	18
10	20 X 24	8
11	19 X 20	8
12	19 X 20	12
13	19 X 20	12
14	19 X 20	12

HOUSTON LEARNING CENTER

South Building

Classroom Number	Size in Feet	Student Capacity
123	20 X 23	18

HOUSTON LEARNING CENTER

North Building

Classroom Number	Size in Feet	Student Capacity
4	16 X 18	10
6	16 X 18	10
8	16 X 18	10
1	18 X 18	10

TYPE OF AIRCRAFT

HS 125/800 series.

CATEGORY OF TRAINING

Initial and Transition training for a HS-125 type rating added to an existing pilot certificate or the issuance of an Airline Transport Pilot Certificate with a HS-125 type rating.

DUTY POSITION

Pilot-in-Command (PIC)

CURRICULUM TITLE

HS 125/800 XP Pilot Training Course and Advanced Simulation Training Program.

CURRICULUM PREREQUISITES

A pilot may enroll in the initial course and complete all of the items of the practical test required for a HS-125 type rating that is authorized to be accomplished in the Level A through Level C simulator, then complete the items required in flight in a HS-125 airplane, if the pilot:

1. Holds a valid Private or Ccommercial Pilot certificate with an Instrument Rating or an Airline Transport Pilot certificate with an Airplane Rating or is a U.S. military appointed Pilot in Command of an airplane of the same class with appropriate written tests passed.

2. Holds an airplane multiengine land rating.

3. Holds a current medical.

A pilot who meets the above may apply for an Airline Transport Pilot certificate with a HS-125 rating provided the pilot:

1. Holds a current first class medical.

2. Holds a F.A.A. Commercial Pilot certificate, or is a U.S. military appointed Pilot in Command, an ICAO recognized Airline Transport Pilot or Commercial Pilot certificate without restrictions.

3. Meets the eligibility requirements of FAR 61.151.

4. Has passed the written test required by FAR 61.153.

5. Meets the experience requirements of FAR 61.155.

Transition Training Prerequisites

A pilot may enroll in the transition course and complete 100% of the practical test required for a HS-125 type rating in an approved Level C simulator, except for the preflight inspection which must be completed in either a static airplane or by using an approved pictorial means, if the pilot:

1. Holds a valid Private or Commercial Pilot certificate with an Airplane and Instrument Rating or an Airline Transport Pilot certificate with an airplane rating, or is a U.S. military appointed Pilot in Command of an airplane in the same class with appropriate written tests passed.
2. Holds an Airplane Multiengine Land Rating.
3. Holds a current medical certificate.
4. Meets one of the following additional prerequisites:
    - a. Holds a type rating in a multiengine turbojet airplane; or
    - b. Has been appointed by a military service as a pilot in command of a turbojet multiengine land airplane; or
    - c. Has at least 2,000 hours of actual flight time, of which 500 hours must be in turbine- powered multiengine airplanes; or
    - d. Has at least 500 hours of actual flight time in a HS-125 series airplane; or
    - e. Has at least 1,000 hours of flight time in at least two different airplanes requiring a type rating; or
    - f. Subject to the limitation FAR 61.64:
        - (1) Holds a type rating in a propeller-driven airplane; or
        - (2) Has, since the beginning of the preceding 12 calendar months, logged at least 100 hours of flight time in multiengine airplanes and at least 25 hours of the flight time were in the HS-125 series airplanes.
        - (3) A pilot qualifying under this subparagraph may not act as PIC of HS-125 series airplane and will be issued a HS-125 type rating or an ATP certificate with a HS125 type rating, as appropriate, with the limitation, "This certificate is subject to pilot-in-command limitations for the added rating," until:
            - (a) The pilot has served 15 hours of Supervised Operating Experience (SOE) as pilot in command of a HS-125 airplane under the supervision of a qualified and current PIC of a HS-125 series airplane.

              AND
            - (b) The pilot has presented evidence of the SOE time to any FAA district office for removal of the limitation; or
    - g. If an applicant does not meet the additional prerequisites of 4a through 4f, he/she may, on completion of the course, receive a certificate with the following limitations:

A pilot qualifying under this subparagraph may not act as PIC of a HS-125 airplane and will be issued a HS-125 type rating or an ATP certificate with a HS-125 type rating as appropriate, with the limitation, "This certificate is subject to pilot-in-command limitations for the added rating," until:

(1) The pilot has served 25 hours of Supervised Operating Experience (SOE) as pilot in command of a HS-125 airplane under the supervision of a qualified and current PIC of a HS-125 airplane.

AND

(2) The pilot has presented evidence of the SOE time to any FAA district office for removal of the limitation.

The PIC limitation is not required if the pilot performs the following on a static airplane or in flight as appropriate:

1. Preflight Inspection
2. Normal Takeoff
3. Normal ILS Approach
4. Missed Approach, AND
5. Normal Landing

A pilot who meets the transition prerequisites above may enroll for an Airline Transport Pilot certificate with a HS-125 type rating, provided he/she meets the following additional requirements:

1. Holds a current first-class medical certificate.
2. Holds a Commercial Pilot certificate or an ICAO-recognized Airline Transport Pilot or Commercial Pilot license without restrictions.
3. Meets the eligibility requirements of FAR 61.151.
4. Has passed the written test required by FAR 61.153.
5. Meets the experience requirements of FAR 61.155.

FLIGHTSAFETY TRAINING POLICY

The policy is to train to "Proficiency" based on need-to-know information.

DESCRIPTION OF INITIAL EQUIPMENT COURSE

The HS-125 Initial Equipment Course is scheduled for fourteen days and consists of the following programmed hours:

Ground Training

FlightSafety Administration	0.5
Aircraft Systems	38.5
Systems Review	2.0
Examination	1.0
System Integration	§0.0
Prebrief/Post-brief (Simulator, LOFT, Aircraft)	12.0
Oral Exam and Pre/Post Briefings for Qualification	3.0
Total Ground Training Hours	57.0

§ Cockpit Procedure Mockup (CPM) training may be given in conjunction with the simulator prebriefs.

Flight Training

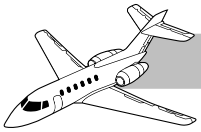
Aircraft Flight Training (Typical)(if necessary)	12.0
Flight Training (LOFT)	4.0
Total Flight Training	16.0

Qualification Check

Flight Simulator Qualification Check (85%)	2.0
Aircraft Check (15%) (If Req'd)	0.5
Total Qualification Hours	Total 2.5

An applicant may choose to take the entire practical test in the aircraft rather than in the simulator.

This course meets the conditions and limitations of FAA Exemptions 5241 and 5317 as amended.



DESCRIPTION OF TRANSITION COURSE

The HS-125 Transition Course is scheduled for approximately fourteen days and consists of the following programmed hours:

Ground Training

Flight Safety Administration	0.5
Aircraft Systems	38.5
Systems Review	2.0
Examination	1.0
System Integration	§0.0
Prebrief/Postbrief (Simulator, LOFT, Aircraft)	12.0
Oral Exam and Pre/Post Briefings for Qualification	3.0
Total Ground Training Hours	57.0

Flight Training

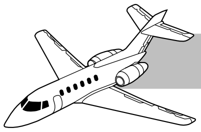
Flight Training (Simulator)	10.0
Aircraft Flight Training (Typical)(if necessary)	2.0
Flight Training (LOFT)	4.0
Total Flight Training Hours	18.0

Qualification Check

Flight Simulator Qualification Check (100%)	2.0
Aircraft Check (if required)	0.5
Total Qualification Hours	2.5

The 12 hours of programmed flight simulator training are for left-seat training. There will be an additional 12 hours of right seat training Normally a pilot is trained as a crew with another pilot. A pilot training not as a crew, will receive the entire LOFT from the left seat.

This course meets the conditions and limitations of FAA Exemptions 5241 and 5317 as amended.



COURSE OBJECTIVES

Upon the completion of this course, the pilot will have the necessary knowledge and skills to: demonstrate that he/she is the master of the aircraft, with the successful outcome of a procedure or maneuver ***never*** in doubt, and meet or exceed the requirements/standards listed in FAA-S-8081-5 Airline Transport Pilot and Type Rating Practical Test Standards.

Successful completion of the HS-125/800 XP Pilot Training Coursewill satisfy the requirements for the following:

- An Instrument Competency Check as specified in FAR 61.57
- Recent Flight Experience as specified in FAR 61.57 (c) and/or 61.57 (d), provided the candidate meets the requirements of FAA Exemption 5317 as amended.
- Pilot-in-Command Proficiency Check as specified in 61.58
- Initial/Transition training for FAR 135 certificate holders contracting with FlightSafety under Exemption 5241.

TRAINING SCHEDULE (TYPICAL)

Listed below is a typical schedule for the pilot training curriculum. On occasion, the schedule may be rearranged to meet the needs of the client or Center. The schedule consists of 11 training days.

		HOURS
Day 1	FlightSafety Administration	0.0
	Classroom	6.0
	Introduction	
	Aircraft General	
	Electrical Power Systems	
Day 2	Classroom	6.0
	Electrical	
	Lighting	
	Master Warning System	
	Fuel	

	HOURS
Day 3 Classroom	6.0
Auxiliary Power Unit	
Powerplant	
Fire Protection	
Pneumatics	
Air Conditioning	
Day 4 Classroom	6.0
Pressurization	
Hydraulic Power Systems	
Landing Gear and Brakes	
Day 5 Classroom	6.0
Flight Controls	
Avionics	
Miscellaneous	
Ice and rain Protection	
Day 6 Simulator — Period No. 1	2.0
Day 7 Simulator — Period No. 2	2.0
Day 8 Classroom	3.0
Performance	
Weight and Balance	

	HOURS
Day 9 Classroom	3.0
Performance	
Simulator — Period No. 4	1.0
Day 10 Classroom	3.0
Performance	
Weight and Balance	
Day 11 Simulator — Period No. 5	2.0
Classroom	3.0
Review and Final Exam	
Day 12 Simulator — Period No. 6	2.0
Day 13 Simulator — Period No. 7	4.0
LOFT (IAW AC 120-35B)	
Day 14 Simulator — Period No. 8	2.0
Simulator Check	

SIMULATOR AND FLIGHT TRAINING

Pilot performance during simulator and flight training shall be graded as: Proficient (1), Normal Progress (2), Additional Training Required (3), Unsatisfactory (4), or Discussed (D).

The criteria for grading shall be as follows:

- Proficient (1) — The pilot is able to easily perform the procedure or maneuver and is the obvious master of the aircraft, with the successful outcome of the maneuver never in doubt.
- Normal Progress (2) — The pilot is making satisfactory progress toward proficiency in the procedure or maneuver but still requires assistance from the instructor. However, the instructor is satisfied that, with additional practice as provided in the standard syllabus, the pilot will become fully proficient in the maneuver or procedure.
- Additional Training Required (3) — The pilot's progress is not satisfactory. However, the instructor is of the opinion that additional training over and above that specified in the syllabus will enable the pilot to become proficient.
- Unsatisfactory (4) — The pilot shows basic deficiencies, such as lack of knowledge, skill, or ability to perform the required procedures or maneuvers. If the present level of performance or progress is maintained, it is doubtful that the pilot will become proficient. Further training shall be given only after review by the Center Manager.
- Discussed (D)—This designation indicates that the item was discussed and not performed in the simulator or aircraft. The discussion revealed a satisfactory knowledge of the appropriate procedure, aircraft system, etc.

Flight Training–Simulator — The pilot is required to achieve a grade of 1 (proficient) by the completion of simulator training. Additional training will be provided in the portion of the flight in which the pilot experienced difficulty. Decision to terminate training for a pilot who demonstrates substandard performance will be made by the Center Manager.

Flight Training–Airplane — The pilot is required to achieve a grade of 1 (proficient) by the completion of flight training. Additional training will be provided in the portion of the flight in which the pilot experienced difficulty. Decision to terminate training for a pilot who demonstrates substandard performance will be made by the Center Manager.

NOTE

Flight time may vary due to weather, air traffic conditions and availability of aircraft.

COMPLETION STANDARDS

The pilot must demonstrate through written examination and simulator/flight practical tests that he/she meets the qualification standards for each segment of the course:

- Aircraft Ground Training — The pilot must demonstrate adequate knowledge of theHS-125 to pass a closed-book systems final exam with minimum passing grades of 70% corrected to 100%.
- Simulator/Flight Training — The pilot will meet the standards of the Airline Transport Pilot and Type Rating Practical Test Standards (FAA-S-8081-5).

TRAINING MATERIALS, AIDS, DEVICES, AND METHODS

Listed below are the training tools to implement the curriculum.

Training materials:

- FlightSafety's HS-125/800 XP Pilot Training Manual
- Manufacturer's Approved Checklist
- FAA-approved HS-125/800 XP Flight Manual
- HS-125/800 XP Crew Manuals
- FlightSafety Pilot's Cockpit Checklist
- Honeywell FMZ-800 FMS Manual
- Honeywell HS-125/800 SPZ-8000 Pilot's Manual

Training aids:

- Computer Graphics Display — Animated Classroom Presentation Systems (ACPS)
- 35mm projector and slides
- Video Presentations
- Overhead projector and viewgraphs
- Liquid chalkboards
- Cockpit panel posters

SPZ-8000 Trainer:

The SPZ-8000 Trainer permits HS-125/800 XP flight crews to become proficient in the use of the Honeywell Integrated SPZ-8000 Digital Automatic Flight Control System. Only the pilot's side of the instrument panel is active.

Cockpit familiarization and interactive student training is possible within the scope of the trainer.

This includes takeoff, climb, cruise, descent, approach, go-around, and landing utilizing the EFIS, FMS and FCS.

The trainer provides total free-play operation and canned flight plans.

Cockpit Procedures Mockup:

The HS-125/800 XP Cockpit Procedures Mockup (CPM) is a non-operational representation of the aircraft cockpit. Photographs of the actual system switch panels are installed for discussion of systems operation. Nonfunctional displays representing flight and engine instruments, circuitbreaker panels, and avionics equipment are provided to aid identification and location of equipment. The CPM is used to introduce the student to the switch and indicator panel layout of the cockpit. It is also used to practice normal and abnonnal/emergency procedures.

Flight Simulator:

The equipment simulates the operational characteristics of the actual aircraft. The simulator flight compartment is a replica of the cockpit. The simulator provides the student with a realistic aircraft platform in which to train in various instrument meteorological conditions, to practice the entire gamut of emergency and abnormal procedures, and to gain confidence and understanding in the HS-125/800 XP aircraft. The control of the simulator is from the instructor's station located behind the flight compartment. From the instructor's station, the instructor controls the aircraft's conditions, the atmospheric and environmental situations, the execution of aircraft system malfunctions, and the simulation of air-to-ground communications. All of these inputs provide the trainee with a realistic flight scenario.

The HS-125 simulators listed below are certified to the level stated by the National Simulations Program Manager (NSPM). See Chapter 14 for simulator certification documentation.

• Wilmington	(FAA 102)	SN 8263	Level C	6° Motion
• Wilmington	(FAA 363)	SN 8347	Level C	6° Motion
• Houston	(FAA 219)	SN 8288	Level C	6° Motion
• Houston	(FAA 074)	SN 8046	Level A	4° Motion

Inoperative Component Authorization/Limitation:

The use of a simulator with an inoperative component may be approved by the POI on a case-by-case basis, so that maneuvers or procedures that involve the use of an inoperative component will not be authorized. The use of the simulator for other maneuvers and procedures for which it has been previously qualified and approved will be authorized. The authorization to use the simulator with an inoperative component will be valid for a period not to exceed 4 months unless the POI and the NSPM coordinate and approve an extension.

Instructional delivery methods:

- Classroom — Teaching lecture/facilitator
- CPM — Guided discussion, demonstration/performance, practice
- Simulator — Guided discussion, demonstration/performance, practice
- Self-directed learning — Self-study program (independent study)
- SPZ-8000 Trainer—Guided discussion, demonstration/performance, practice

SUPPLEMENTAL TRAINING MATERIAL

All Centers have a self-learning room that contains the following supplemental materials for the student to view.

Computer-based training and interactive videodisc training:

- Practical Cockpit Resource Management
- Computer-based training for the HS-125/800 XP SPZ-8000 System

Video Tape General Topics Areas

- Flight
- Windshear
- Navigation and radar systems
- Physiology
- Weather
- Regulations and procedure
- HS-125/800 XP aircraft preflight
- HS-125/800 XP FMS operation

NOTE

The review and final examination and performance classroom segments may be varied in the order presented. Simulator periods 7 and 8 may be given in either order.

PERSONNEL QUALIFICATIONS

QUALIFICATIONS—MANAGEMENT

Manager

The Manager is responsible for operation of the Learning Center, including planning, staffing, administration, and marketing. The manager should have a college degree or equivalent and have five years' experience in management or sales at a decision-making level.

Assistant Manager

The Assistant Manager will assume responsibility of the Learning Center in the absence of the Manager. The Assistant Manager should have a college degree or equivalent and have two years' experience in management or sales at a decision-making level.

Director of Pilot Standards

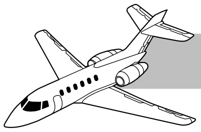
The Director of Pilot Standards is responsible for the review of all technical publications, flight training manuals, and other materials to ensure compliance with the approved Flight Manual limitations and procedures, the FlightSafety Learning Center Operations Manual, and the Federal Aviation Administration regulations and directives, including current exemptions. The Director of Pilot Standards is responsible for evaluation, training and documentation of required training for all simulator and flight instructor personnel. The Director of Pilot Standards should have an Airline Transport Pilot Certificate with appropriate type ratings and a Flight Instructor Certificate with instrument and multiengine ratings, three years corporate pilot experience or equivalent, served as a ground and/or flight instructor for 2 years, and have at least two years of college or equivalent.

Director of Pilot Training

The Director of Pilot Training is responsible for developing and instituting departmental training polices and procedures, and integrating Company and Standards Division policies. This includes the preparation of short- and long-range training plans, curricula and syllabi, training delivery, condition of training devices, scheduling, and documentation. The Director of Pilot Training should have an Airline Transport Pilot Certificate with appropriate type ratings and a multiengine rating, five years corporate pilot experience or equivalent, two years related aviation management experience, served as a ground, simulator, and/or flight instructor for a minimum of two years, along with at least two years of college or equivalent.

Program Manager

The Program Manager is responsible for supervising and managing the day-to-day client training activities specific to the program aircraft to include aircraft-specific training delivery, client scheduling, condition of training equipment, and training documentation. The Program Manager coordinates with the lead center in formulating and standardizing the satellite learning center's training programs, other Program Managers to facilitate standardization, acts as a ground simulator and/or flight instructor, acts as an FAA Designated Pilot Examiner (DPE), Pilot Proficiency Examiner (PPE), FAR 121/135 Check Airman, Designated Simulator Examiner, and/or a School Designated Examiner (SDE) as required, and ensures compliance with all regulations and di-



rectives. The Program Manager must possess an Airline Transport Pilot Certificate with a Multi-engine Rating and a type rating for the program aircraft, a Certified Flight Instructor Certificate with Instrument and Multi-engine Ratings, three years corporate pilot training experience or equivalent commensurate with the typical program client, a minimum of one year aviation management experience, served as a ground, simulator, and/or flight instructor for a minimum of one year, and have two years college or equivalent.

QUALIFICATIONS—PILOT GROUND SCHOOL INSTRUCTORS

Each Ground School Instructor assigned to this course must hold a Ground Instructor/Advanced Instrument or Flight Instructor Certificate and complete initial and recurrent training.

QUALIFICATIONS—FLIGHT AND/OR SIMULATOR INSTRUCTORS

Each flight instructor assigned to this course must meet the requirements of FAR Part 61, Subpart G, and hold a Flight Instructor Certificate with Airplane Instrument and Multiengine Land ratings and an Airline Transport Pilot Certificate with a CE-500 type rating.

Each Simulator Instructor must meet the requirements of FAA Exemption 5317, as amended.

Initial training shall consist of at least the following:

- FlightSafety Instructor Development Recurrent Course
- One initial pilot ground school course
- Six hours observation of qualified simulator instruction
- Performance as an instructor under the observation of the Director–Pilot Training or designee for a period of time to be determined by the Director–Pilot Training
- Endorsement of Training Record for Instruction by the Director–Pilot Training
- Review of pertinent information relative to flight/simulator instructor duties: instructor guides/presentation plans, syllabus, FARs, etc.
- For flight instructors, completion of at least one physiological training course
- For flight instructors, observation of a qualified CFI instructing in an aircraft for a complete flight syllabus
- Receive a flight instructor check by the Director–Pilot Training or designee
- View the videotapes on windshear as listed in the FlightSafety Video Catalog

Recurrent training for simulator instructors shall be conducted semiannually and consist of at least the following:

- Director–Pilot Training or designee will observe and evaluate the instructor to determine that FlightSafety teaching standards are met. Evaluation will be documented in the Instructor Training Record and discussed with the instructor. The evaluation determines the amount of training that is necessary for the instructor.

- Annually simulator instructors will be evaluated in accordance with Exemption 5317 as amended. See Chapter 8 for details of initial and recurrent training for ground, simulator, and flight instructors.

Current FlightSafety Instructors may qualify to take the FAR 61.58 PIC Check in the HS-125 Simulator without completing the Chapter 10 syllabus (See TAB 7) provided the instructor has piloted the HS-125 Simulator at least 3 hours in the 180 days preceding the date of the FAR 61.58 PIC Check.

Flight instructors must have in addition to the preceding:

- One of the semiannual simulator observation-evaluations conducted in the aircraft (if possible)

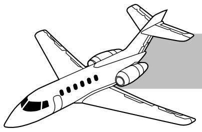
- Annual evaluation while instructing in each type of aircraft simulator in which the instructor conducts training

MISCELLANEOUS

RECORD-KEEPING REQUIREMENTS

Student and instructor records will be kept on file in the Learning Center administrative area. Student Training Records will consist of at least the following:

- Full name of pilot
- Pilot’s aeronautical experience
- Total flight time at the beginning of course
- Total flight time in the airplane type
- Pilot certificate
- Current medical certificate
- Ground school training record
- Results of aircraft ground school written examination (answer sheet)
- FlightSafety graduation certificate (diploma)
- FlightSafety Record of Training
- Flight Training Record/Simulator Training Record



Instructor Training Records will consist of at least the following:

- Resume of pilot at time of employment
- Ground school training records
- FlightSafety graduation certificate (diploma)
- FlightSafety Record of Training
- Flight Training Record/Simulator Training Record
- Landing Currency Certificate
- A copy of Instructor Development Course diploma
- Copies of the following certificates issued by outside agencies:
    - Pilot Certificate
    - Fight Instructor Certificate
    - Ground School Instructor Certificate (if applicable)
    - Current medical certificate
    - Examiner designation (if applicable)
    - Pilot Proficiency Examiner (PPE) authorization (if applicable)
- Copies of the following certificates issued by FlightSafety:
    - Instructor Certification
    - Academic and/or Flight/Simulator Instructor Evaluations
    - Academic Instructor Certification (if applicable)
    - Simulator/Flight Instructor Certification and Training Record (if applicable)

Instructors and pilot check airmen who are authorized to instruct or check for FAR Part 135 operators will have the following additional information, required by FAR Part 135.63(a)(4), in their training record:

- Full name of the pilot
- Pilot certificates (by type and number) and ratings that the pilot holds
- Pilot's aeronautical experience in sufficient detail to determine the pilot's qualification to pilot aircraft in operations under this part

- Pilot's current duties and the date of the pilot's assignment to those duties
- Effective dates and class of the medical certificates that the pilot holds
- Date and results of each of the initial and recurrent competency tests and proficiency and route checks required by this part and the type of aircraft flown during that test or check
- Pilot's flight time in sufficient detail to determine compliance with the flight time limitations of this part
- Pilot's check pilot authorization, if any
- Any action taken concerning the pilot's release from employment for physical or professional disqualification
- Date of the completion of the initial phase and each recurrent phase of the training required by this part

TESTING FOR PROHIBITED DRUGS

FlightSafety International's drug testing program as mandated by the Federal Aviation Administration commenced as of December 11, 1990. Gregory and Howe, Inc., has been contracted to administer this program.

In addition to the drug testing program, FlightSafety offers an Employee Assistance Program (EAP). The EAP provides education and training programs for all employees. Education programs are in the form of display and distribution of material on community service hot-line telephone numbers and FlightSafety's policy regarding drug use in the work place. All employees receive initial training on the causes and consequences of using drugs and the signs that someone is using drugs. Additionally, supervisory personnel receive training on specific, contemporary physical, behavioral, and performance indicators of probable drug use.

AIRCRAFT GROUND TRAINING CURRICULUM

CURRICULUM SEGMENT OUTLINE

1. Aircraft Ground Training Curriculum Segment
    A. Objective: To provide pilots with the necessary knowledge for understanding the basic functions of aircraft systems, the use of the individual system controls, and the integration of aircraft systems with operational procedures to sufficiently prepare them to enter the flight training curriculum segment.
    B. FlightSafety Administration
    C. General Operational Subjects Modules (if applicable)
    D. Aircraft Systems Modules
        1. Aircraft General
        2. Electrical
        3. Lighting
        4. Master Warning
        5. Fuel
        6. Auxiliary Power Unit (APU)
        7. Engines
        8. Fire Protection
        9. Pneumatics
        10. Air Conditioning
        11. Pressurization
        12. Ice and Rain Protection
        13. Hydraulics
        14. Thrust Reversers
        15. Landing Gear and Brakes
        16. Flight Controls
        17. Avionics
        18. Oxygen
        19. Aircraft Performance
        20. Weight and Balance
        21. Flight Planning

E. Systems Integration Modules
   1. Aircraft Preflight Walkaround
   2. SPZ-8000
   3. Cockpit Procedures Mockup
   4. Crew Resource Management

F. Systems Review

G. Examination and Critique

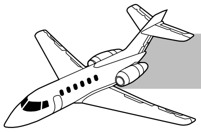
H. Completion Standards

   1. Systems—The pilot must demonstrate adequate knowledge of the aircraft systems by passing an open-book, written final exam with a score of 70% that is corrected to 100%.

GROUND SCHOOL TRAINING HOURS SUMMARY

	Hours
FlightSafety Administration	0.5
Aircraft Systems	38.5
Systems Review	2.0
Examination	1.0
Pre-brief/Post-brief (Simulator, CPM, LOFT, Aircraft)	12.0
Oral Exam and Pre/Post briefings for Qualification	3.0

Total 58.5 Minimum to 61.5



TRAINING MODULE OUTLINES

General Operational Subjects

The subject of ground training, referred to as "General Operational Subjects," includes instruction on certain operational requirements that are specific to the FAR 135 certificate holder and to the aircraft in which the training is being conducted. Training in general operational subjects is not conducted by FlightSafety unless specifically pertinent to this course.

I. Aircraft Systems Modules
  A. Aircraft General/Master Warning Module
    1. General Elements
      a. System Description
        (1) Structures
        (2) Airplane Systems
        (3) Walkaround
      b. Controls and Components
        (1) Control Locks
        (2) Covers, Plugs, and Pins
        (3) Davtron Clocks
        (4) Dim Override
        (5) Glareshield Panels
        (6) Hand-Held ~BCF Extinguisher
        (7) Fire Ax
        (8) Harness and Windows
        (9) MWS TEST
        (10) Overhead Panel
        (11) Test annunciator lights
      c. Servicing/Preflight
        (1) AFM
        (2) Documents:
          (a) Airworthiness Certificate
          (b) Registration Certificate
          (c) Weight and Balance Manual
          (d) Airplane Radio Station License
          (e) Exit and Emergency Lights

        - (4) Overhead Panel
        - (5) Two flashlights
    - 2. Limitation Elements
        - a. Limitations
            - (1) Design Weights
            - (2) Types of Operations
            - (3) Maximum Altitude
            - (4) Maximum Crosswind Component
            - (5) Maneuver Limitations
            - (6) Maximum Occupancy
            - (7) Airspeed Limitations
            - (8) Bird Strike Protection
- B. Electrical Module
    - 1. General Elements
        - a. System Description
            - (1) General
            - (2) DC Power
            - (3) AC Power
            - (4) Electrical Power Sources
            - (5) Limitations
        - b. Controls and Components
            - (1) DC Power
                - (a) Battery Isolation Switch
                - (b) Batteries 1, 2, 3, 4, and 5
                - (c) Battery Switch
                - (d) Bus Tie
                - (e) Bus Tie Switch
                - (f) External Battery Charge Switch
                - (g) External Power Switch
                - (h) Generator Switch
                - (i) Generator Ammeter
                - (j) Generator Line Contactor (GLC)
                - (k) PS Bus Bars
                - (l) PE Bus
                - (m) Radio Masters/EFIS Switch

- (n) START PWR Switch
- (o) Circuit Breakers
  - 1) White Background
  - 2) Red Background
- (2) AC Power
  - (a) Alternator
  - (b) Inverters
  - (c) Standby Inverter
  - (d) STBY INV Switch
  - (e) XE Bus
  - (f) Circuit Breakers
    - 1) XE-supplied circuit breakers
    - 2) STBY INV
    - 3) XE Bus Supply
- c. Indicators/Indications
  - (1) Battery and Generator Ammeters
  - (2) BATT HOT warning
  - (3) Battery Charge
  - (4) Battery Temperature
- d. Annunciators
  - (1) Caution [Warning Panel
    - (a) ALTR 1 FAIL
    - (b) ALTR 2 FAIL
    - (c) BATT CNTCTR
    - (d) BATT HOT
    - (e) BATT O/HEAT
    - (f) BUS TIE OPEN
    - (g) E BATT ON
    - (h) GEN 1 FAIL/GEN 2 FAIL
    - (i) INST 1 FAIL/INST 2 FAIL
    - (j) INV 1 FAIL/INV 2 FAIL
    - (k) STBY INV ON
    - (1) XE FAIL
    - (m) XS l FAIL
    - (n) XS 2 FAIL

- e. Servicing/Preflight
    - (1) Battery Isolate Switch
    - (2) Battery Switch
    - (3) External Battery Charge Switch
- 2. Operational Elements
    - a. Battery Capacity Check
    - b. Check Battery and Fuel Temperature
    - c. Check Engine Instrument and Electrical System
- 3. Limitation Elements
    - a. Minimum Battery Voltage
    - b. Generator Limits
    - c. Ground Power Unit Limits
- 4. Emergency/Abnormal Procedure Elements
    - a. Abnormal Voltage
    - b. Battery Contactor
    - c. Battery Overheat
        - (1) Battery Hot
        - (2) XE Fail
        - (3) XS1 Fail and XE Fail
        - (4) XS1 or XS 2 Fail
    - d. Double Generator Failure
    - e. Double Inverter Failure
    - f. Emergency Battery Supply
    - g. Generators Out Of Balance
    - h. Single Generator Failure
    - i. Single Inverter Failure
    - j. Voltage Fluctuation
    - k. Annunciators from I. B. 1. d.
    - l. Abnormal Electrical Conditions
    - m. BUSTIE Open

C. Lighting Module
  1. General Elements
    a. System Description
      (1) Interior Lighting
      (2) Exterior Lighting
      (3) Electrical Power Sources
    b. Controls and Components
      (1) Aisle Lights
      (2) Beacon/Pulse
      (3) Cabin signs on
      (4) Cabin Lights
      (5) Cabin Emergency Lights
      (6) Cockpit Emergency and Storm Lights
      (7) Emergency Lights
      (8) Exterior Lights
      (9) Instrument Panel Lights
      (10) Interior Lights
      (11) Landing Lights
      (12) Logo Lights
      (13) Taxi Lights
      (14) Lavatory Lights
      (15) Navigation Lights
      (16) No Smoking Switch
      (17) No Smoking and Seat Belt Signs
      (18) Panel Lights
      (19) Rotating Beacon
      (20) Strobe Lights
      (21) Taxi Lights
      (22) Wing Inspection Lights

D. Master Warning Module
  1. General Elements
    a. System Description
      (1) Visual Indications
      (2) Audible Indications

b. Controls and Components
(1) Center Annunciator Panel
(2) Roof Annunciator Panel
(3) Test Panel
c. Indicators/Indications
(1) Master Warning Lights
(2) Engine Fire Detection Audible Warning
(3) Landing Gear Audible Warning
(4) Airbrakes Horn Warning
(5) Throttle Below 60%-70% rpm
(6) Speed Warning
(7) Testing
(8) Cabin Depressurization
(9) Airframe De-icing Audible Warning
2. Operational Elements
a. Master Warning System Operation
b. Dim Override Test
c. Smoke Detector Test
E. Fuel Module
1. General Elements
a. System Description
(1) Fuel Tanks
(2) Fuel Distribution
(3) Controls and Indicators
(4) Refueling
(5) Electrical Power Sources
(6) Limitations
b. Controls and Components
(1) Auxiliary Transfer Lever
(2) Fuel Pumps
(3) Fuel Temperature Button
(4) HP Fuel Cock
(5) LP Fuel Cock
(6) Wing Fuel Crossfeed and Transfer Lever

c. Indicators/Indications
  (1) Fuel Counter
  (2) Fuel Flow
  (3) Fuel Quantity
  (4) Fuel Temperature
  (5) REFUEL ON Light
d. Annunciators
  (1) ENG 1 LO PRESS/ENG 2 LO PRESS
  (2) WING FUEL XFD/TFR
  (3) AUX FUEL XFER
e. Servicing/Preflight
  (1) Fuel Quantity
  (2) No. 1 and No. 2 Fuel Pumps
  (3) Refuel Panel
2. Operational Elements
  a. Check Battery and Fuel Temperature
  b. Maintain a lateral balance of fuel
3. Limitation Elements
  a. Distribution in Wing Tanks
  b. Transfer from Ventral Tank to Wing tanks
  c. Landing with fuel in Ventral Tank
4. Emergency/Abnormal Procedure Elements
  a. Auxiliary Fuel Transfer Failure
  b. Fuel Low Pressure
  c. Annunciators from 1. E. 1. d.

F. APU (Auxiliary Power Unit) Module
  1. General Elements
    a. System Description
      (1) Garrett APU (GTCP 36-150W)
      (2) Turbomach APU (T62T-40)
      (3) Electrical Power Sources
      (4) Limitations

b. Controls and Components
   - (1) APU Master Switch
   - (2) APU Fire Light
   - (3) APU Fire Extinguisher Switch
   - (4) APU Generator Switch
   - (5) Fire Warning Test Button
   - (6) Rear Equipment Bay
   - (7) Start/Run/Stop Switch
   - (8) Over Voltage Trip Switch

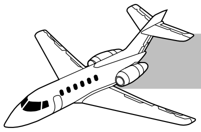
c. Annunciators
   - (1) APU FAULT
   - (2) APU FIRE
   - (3) APU GEN OVERHEAT
   - (4) APU WARN
   - (5) LO OIL PRESS
   - (6) HIGH OIL TEMP
   - (7) APU GEN
   - (8) BLEED AIR VALVE FAIL
   - (9) BLEED AIR ON
   - (10) READY TO LOAD

d. Servicing/Preflight

2. Operational Elements
   a. Start the APU

3. Limitation Elements
   a. Inflight Operation of APU
   b. Normal Electrical Load
   c. Starting Limitations
   d. Maximum Altitude
   e. Maximum Starting Airspeed
   f. Generator Load Limit

4. Emergency/Abnormal Procedure Elements
   a. APU Failure
   b. APU Fire
   c. Annunciators from 1. F. 1. c.
   d. APU Generator Overheat
   e. APU Generator Failure



G. Engine Module

1. General Elements
    a. System Description
        (1) Major Sections
        (2) Engine Systems
        (3) Engine Instrumentation
        (4) Engine Power Control
        (5) Engine Starting
        (6) Engine Synchronization
        (7) Automatic Performance Reserve (APR)
        (8) Airstarts
        (9) Electric Power Sources
        (10) Power Plant Limitations
    b. Controls and Components
        (1) APR
        (2) APR ARM Selector Switch
        (3) APR O/RIDE Selector Switch
        (4) ENG FIRE Light
        (5) ENG IGNITION 1 and 2 Switches
        (6) Engine Computer Light
        (7) Engine Computers
        (8) Engine Instruments
        (9) Engine Start Power Switch
        (10) Engine Start Switch
        (11) Engine Synchronizer
        (12) HP Cock
        (13) LP Cocks
        (14) Master warning lights
        (15) Power Levers
        (16) Thrust Reverser

c. Indicators/Indications
  - (1) HP Cock Fire Light
  - (2) Engine Instruments
  - (3) APR lights
  - (4) ITT
  - (5) N_1 rpm
  - (6) N_2 rpm
  - (7) Oil Press Lights
  - (8) Arm/Off Light
  - (9) Unlocked Light
  - (10) Reverser Light

d. Annunciators
  - (1) ENG 1 CMPTER
  - (2) ENG 2 CMPTER
  - (3) START PWR
  - (4) L-R OIL PRESS
  - (5) Green Operating lights

2. Operational Elements
   - a. Aurally check ignitor plug operation
   - b. Check APR
   - c. Check Engine Computer/Rudder Bias/Thrust Reversers
   - d. Check Engine Instrument and Electrical System
   - e. Check Oil Level and Filter
   - f. Start Engines
3. Limitation Elements
   - a. Start Operating Lights
   - b. Operating Limits
   - c. Oil Pressure and Temperature
   - d. Approved Oil
   - e. Oil Servicing

4. Emergency/Abnormal Procedure Elements
    a. Abnormal Engine Indications Or Behavior
    b. Engine Compressor Stall
    c. Engine Computer Malfunction
    d. Engine Fire In Flight
    e. Engine Fuel Malfunction
    f. Engine Fire Warning On The Ground
    g. Engine Failure
    h. Fuel Management After One Engine Failure
    i. Oil Low Pressure
    j. Immediate Relight
    k. Starter Assisted Relight
    1. Windmill Relight
    m. Annunciators from I.G.I.d.
    n. High Engine Vibration
    o. Rudder Bias Not Inhibited
    p. Unlock/or Arm In Flight
    q. Autostow
    r. Double Engine Failure

H. Fire Protection Module
  1. General Elements
    a. System Description
      (1) Fire Detection
      (2) Fire Extinguishing
      (3) Electrical Power Sources
    b. Controls and Components
      (1) APU Fire Extinguisher Switch
      (2) ENG FIRE 1
      (3) ENG FIRE 2
      (4) Fire Extinguisher
    c. Indicators/Indications
      (1) Fire Extinguisher Indicators/Warning Bell
    d. Annunciators
      (1) FIRE 1-Isolate Bell/FIRE 2-Isolate Bell
      (2) FIRE 1 Test/FIRE 2 Test

e. Servicing/Preflight
(1) Preflight Extinguishers
(2) Check Discharge Discs
2. Operational Elements
a. Discharge Extinguishers to Fight Fire
3. Emergency/Abnormal Procedure Elements
a. APU Fire
b. Annunciators from 1. H. 1. d.
I. Pneumatics Module
1. General Elements
a. System Description
(1) Description and Operation
(2) Electrical Power Sources
(3) Limitations
b. Controls and Components
(1) Flight Deck Heat Valve
(2) Main Air Valve
c. Annunciators
(1) HP AIR O HEAT
2. Limitation Elements
a. Main Air Valves and Flight Deck Heat Valves for Takeoff and Landing
3. Emergency/Abnormal Procedure Elements
a. Annunciators from 1. 1. 1. c.
J. Air Conditioning Module
1. General Elements
a. System Description
(1) Description and Operation
(2) Electrical Power Sources
(3) Limitations

b. Controls and Components
(1) Cabin Fan
(2) Cabin Flood Flow
(3) Cabin Floor
(4) Cabin Temperature Control Switch
(5) Cockpit Fan
(6) Flight Deck Heat Valve
c. Indicators/Indications
(1) Cabin Temperature
(2) Duct Temperature
d. Annunciators
(1) REAR BAY O'HEAT
2. Emergency/Abnormal Procedure Elements
a. Automatic Temperature Control Failure
b. Duct Overheat
c. High Pressure Air Overheat
d. Annunciators from 1. J. 1. d
K. Pressurization Module
1. General Elements
a. System Description
(1) Operation
(2) Electrical Power Sources
(3) Limitations
b. Controls and Components
(1) Automatic Pressure Controller
(2) Dump Valve
(3) Ground Test Valve
(4) Horn Isolate Switch
(5) Main Air Valves
(6) Manual Pressure Controller
(7) Press Override
c. Indicators/Indications
(1) Cabin Triple Indicator

        - d. Annunciators
            - (1) CABIN ALTITUDE
    - 2. Operational Elements
        - a. Set Auto Press Controller
        - b. Test CABIN ALTITUDE Annunciator
        - c. Verify on the cabin triple indicator that the airplane has pressurized.
    - 3. Limitation Elements
        - a. Maximum Cabin Differential Pressure
        - b. Maximum Allowable Negative Pressure Differential
        - c. Safety Valves Setting
    - 4. Emergency/Abnormal Procedure Elements
        - a. Automatic Pressure Control Failure
        - b. Depressurization
        - c. Emergency Descent
        - d. Failure To Pressurize
        - e. Annunciators from 1. K. 1. d.
- L. Ice and Rain Protection Module
    - 1. General Elements
        - a. System Description
            - (1) Ice Detection System
            - (2) Airframe Deicing System
            - (3) Windshield Anti-Icing System
            - (4) Pitot/Vane Heat System
            - (5) Engine Anti-Ice System
            - (6) Operating Procedures
            - (7) Electrical Power Sources
            - (8) Limitations

b. Controls and Components
  (1) Alternators
  (2) Ammeter Reading
  (3) Engine Anti-Ice Switches
  (4) Ice Detect Switch
  (5) Ice Protection
  (6) Pitot/Vane Heat Switches
  (7) Windshield Heat Switches
  (8) Windscreen Alternators
  (9) TKS
  (10) Wing/Tail Anti-ice Time Switch
c. Indicators/Indications
  (1) TKS System Chime
  (2) TKS Gage
d. Annunciators
  (1) ALTR 1 FAIL/ALTR 2 FAIL
  (2) ANTICE LOW PRESS
  (3) ANTICE LOW QUANTITY
  (4) ENG 1 A/ICE / ENG 2 A/ICE
  (5) ICE DETECTED
  (6) ENG ICE PROT ON
  (7) L-R VANE HT FAIL
  (8) PITOT HT FAIL
  (9) L-R SCREEN O’HEAT
  (10) SIDE SCREEN O’HEAT
e. Servicing/Preflight
  (1) TKS Quantity

2. Limitation Elements
  a. Anti-Ice use when not in Icing Conditions
  b. Operation when Ground Cold Soaked
  c. Windscreen failed or turned OFF inflight

3. Emergency/Abnormal Procedure Elements
    a. Alternator Failure
    b. Double Alternator Failure
    c. Side Screen Overheat
    d. Windscreen Overheat
    e. Engine Anti-Ice Failure
    f. Annunciators from 1. L. 1. d.
    g. Pitot Heater Failure
    h. Vane Heater Failure
    i Windscreen Damage

M. Hydraulics Module

1. General Elements
    a. System Description
        (1) Main Hydraulic System
        (2) Auxiliary Hydraulic System
        (3) Hydraulic Subsystems
        (4) Electrical Power Sources
    b. Controls and Components
        (1) Auxiliary Hydraulic System Hand Pump
        (2) Handle Socket
        (3) Pin Indicator
    c. Indicators/Indications
        (1) Hydraulic Tank Pressure
        (2) Main Supply Pressure
    d. Annunciators
        (1) AUX HYD LOW LEVEL
        (2) HYD 1 or HYD 2 LO PRESS
        (3) EMER BRAKE LO PRESS
    e. Servicing/Preflight
        (1) Check Fluid Level
    f. Miscellaneous
        (1) Emergency Accumulator
        (2) Main Accumulator
        (3) Thrust Reverser Accumulator

2. Operational Elements
   a. Reset Auxiliary Hydraulic System Handle
3. Emergency/Abnormal Procedure Elements
   a. Hydraulic Low Pressure
   b. Hydraulic System Overheat
   c. Leak Scenarios
   d. Annunciators from 1. M. 1. d.
   e. Main Hydraulic Failure
   f. EMER Brake Low Pressure
   g. HYD AUX System Low

N. Thrust Reversers Module
1. General Elements
   a. System Description
      (1) Thrust Reverser Assembly
      (2) System Operation
      (3) Electrical Power
      (4) Normal Condition Management
      (5) Abnormal Condition Management
      (6) Limitations
   b. Controls and Components
      (1) Thrust Reverser Levers
      (2) Thrust Reverser Arm Switch
   c. Annunciators
      (1) ARM/OFF
      (2) REVERSER
      (3) REVERSE
      (4) UNLOCK
   d. Servicing/Preflight
      (1) Thrust Reverser
2. Operational Elements
   a. Check Thrust Reversers

3. Limitation Elements
    a. Ground Ops Only
    b. Engine Starts
    c. Deployment on Unpaved Surfaces
    d. Full Reverse Minimum Speed
4. Emergency/Abnormal Procedure Elements
    a. Thrust Reverser Malfunctions In Flight
    b. Annunciators from 1. N. 1. c.

O. Landing Gear and Brakes Module
1. General Elements
    a. System Description
        (1) Landing Gear
        (2) Nosewheel Steering
        (3) Brakes
        (4) Electrical Power Sources
        (5) Limitations
    b. Controls and Components
        (1) Brake Pedals
        (2) Brakes
        (3) Aux Hydraulic Pull Handle
        (4) Gear Handle
        (5) Hand Pump
        (6) Landing Gear
        (7) Landing Gear Lever
        (8) Landing Gear Override
        (9) Landing Gear Warning Horn
        (10) Landing Lights
        (11) Nosewheel Steering Wheel
        (12) Parking Brake
        (13) Pressurization Gear Override Switch
        (14) Three landing gear pins.
        (15) Wheel Brake Lever
        (16) Circuit Breakers
            (a) L/Gear Horn

- c. Indicators/Indications
    - (1) Airbrake Indicator
    - (2) Gear Standby Indicators
    - (3) HYD LO PRESS lights
    - (4) Index should correspond to nosewheel position.
    - (5) Landing Gear Lights
    - (6) Normal Brake Indicator
    - (7) Nose Gear Mechanical Indicator
    - (8) Standby Gear Indicator Lights
    - (9) Zero Supply Gage Reading
- d. Annunciators
    - (1) EMERG BRK LO PRESS
    - (2) HYD LO PRES
    - (3) AUX HYD LOW LEVEL
- e. Servicing/Preflight
    - (1) Brakes
    - (2) Emergency Accumulator
    - (3) Reservoir Air Pressure Gage
    - (4) High Pressure Filter Button
    - (5) Reservoir (2)
    - (6) Landing Gear Handle
    - (7) Main Accumulator

2. Operational Elements
    - a. Check Brakes and Steering
    - b. Set Parking Brake
    - c. Check that the handwheel is clear and the index mark is forward
3. Limitation Elements
    - a. Landing Gear Airspeed Limits (V_{LO},V_{LE})
    - b. Brake Cooling Periods

4. Emergency/Abnormal Procedure Elements
   - a. Emergency Brake Supply Failure
   - b. Failure Of Landing Gear To Retract
   - c. Nosewheel not Centered
   - d. Landing Gear Three Greens Not Indicated
   - d. Normal Brake Supply Failure
   - e. Annunciators from 1. 0. 1. d.

P. Flight Controls Module

1. General Elements
   - a. System Description
     - (1) Primary Flight Controls
     - (2) Trim Systems
     - (3) Secondary Flight Controls
     - (4) Stall Warning and Identification
     - (5) Electrical Power Sources
     - (6) Limitations

b. Controls and Components

- (1) Aileron Trim
- (2) Airbrake Handle
- (3) Airbrakes
- (4) Control Column
- (5) Control Locks
- (6) Elevator Trim
- (7) Elevator Servo
- (8) Flaps
- (9) Flaps Selector
- (10) Hand Pump
- (11) Left and Right Stall Warning Motors
- (12) Manual Trim Wheel
- (13) No. 1, No. 2 and No. 3 Stall Ident
- (14) No. 1, No. 2 and No. 3 Stall Test Button
- (15) Rudder Bias Switches
- (16) Stall Identification Diagnostic Panel
- (17) VLV A Open Lights

(18) VLV B Open lights
(19) Stick Shaker
(20) Stick Pusher
(21) Trim Tabs
(22) Wing stall vents.
(23) Yaw Damper
(24) Circuit Breakers
   (a) Elevator Electric Trim
   (b) Left and Right Stall Warning

c. Indicators/Indications
   (1) Electric Trim Indicator
   (2) Aileron Trim Indicator
   (3) Angle-of-Attack Indicator
   (4) Airbrake Indicator

d. Annunciators
   (1) ELEV/AIL TRIM
   (2) FLAP Fault
   (3) STALL IDENT 1,2 and 3
   (4) SQUAT Fault
   (5) SSU Fault
   (6) RUDDER BIAS

2. Operational Elements
   a. Operate Electric Pitch Trim
   b. Check Flight Controls
   c. Set Flaps
   d. Elevator Trim Check
3. Limitation Elements
   a. Wing Flaps Airspeed Limit (V_{FE})
   b. Use of 15° Flaps for Descent and Holding
   c. Use of Electric Trim
   d. Use of Air-brakes with flaps extended
   e. Use of Lift Dump
   f. Use of Rudder Bias for Takeoff
   g. Resetting Stall Warning and Ident Channels inflight

4. Emergency/Abnormal Procedure Elements
    a. Elevator Trim Runaway
    b. Stick Shaker and/or Pusher Operation
    c. Landing with Elevator Cable Failure
    d. Annunciators from I.P.1.d.

Q. Avionics Module (See note below)
  1. General Elements
    a. System Description
      (1) Pitot-Static System
      (2) FCS-86 Flight Control System
      (3) Electric Power Sources
      (4) Limitations
    b. Controls and Components
      (1) ADC/ADS 1 & 2 Test Switches
      (2) ADC/ADS 1 & 2 Transfer Switches
      (3) AHRS/IRS Panels
      (4) Air Data Transfer Switches
      (5) Altitude Select
      (6) AP DISC Button
      (7) Audio Panels
      (8) Autopilot/AFCS Engage Lever/Switch
      (9) Autopilot/AFCS Pitch Wheel Turn Knob
      (10) Autopilot Sync/TCS Switches
      (11) Autopilot Turn Knob
      (12) EFIS Test Switches
      (13) Comparator
      (14) Comparator Warning Reset Button
      (15) Davtron Clocks
      (16) EFIS Master Switches
      (17) Electric Pitch Trim
      (18) Flight Director/Flight Guidance Panel
      (19) Go-Around Button
      (20) HDG/DG Pushbutton
      (21) LO V_{MO} 1/2 Test Buttons

(22) V_{MO} M_{MO} Test Buttons
(23) Radio Master Switches
(24) Radio Emergency Switch
(25) Pitot Isolation Valve
(26) Remote HDG Selector
(27) Rudder Bias
(28) Standby Altimeter
(29) Standby Horizon
(30) Transfer Panels/Reversionary Mode Panels
(31) Transponder
(32) Yaw Damper

NOTE

EFIS items are taught on an “AS NEEDED” basis in an EFIS/FMS Trainer.

c. Indicators/Indications
(1) Airspeed Indicator
(2) Altimeter
(3) VSI
(4) Standby Horizon
(5) EHSI
(6) TAS Indicator
(7) Altitude Alerter
(8) Attitude Heading Reference System
(9) Autopilot/AFCS
(10) EADI
(11) Fast/Slow Indicator
(12) FMS Messages and Annunciators
(13) MFD
(14) Radar Altimeter
(15) SAT/TAS
(16) V_{MO}/M_{MO}

d. Annunciators
   (1) ALT ALERT
   (2) AUX PWR
   (3) L AFCS FAIL
   (4) R AFCS FAIL
   (5) AP OFF
   (6) AP TRIM (ELV/AIL TRIM)
   (7) Sync/TCS
e. Servicing/Preflight
   (1) Check Radios Transmit and Receive

2. Operational Elements
   a. Test Altitude Alert
   b. Tune and identify all nav systems and radios
   c. Apply aileron and note rudder movements in correct sense.
   d. Check/Set Autopilot/AFCS and Pitch Trim
   e. Check navigation radio reception and bearing information, and set for departure
   f. Check resistance to movement of rudder, aileron, and elevator and appropriate movement of autopilot trim indicators
   g. Disconnect the yaw damper and rudder should be free
   h. Set Nav, Radios and Display Panel
3. Limitation Elements
   a. Maximum speed for Autopilot Operation
   b. Minimum altitude for Autopilot Operation
   c. Use of AFCS/Autopilot or Yaw Damper on takeoff and landing
   d. Use of AHRS for Navigation
4. Emergency/Abnormal Procedure Elements
   a. Air Data Computer Failure
   b. Speed Discrepancy
   c. Reversionary Modes
   d. Annunciators from 1. Q. 1. d.

R. Oxygen Module
  1. General Elements
    a. System Description
      (1) Oxygen Cylinder Assembly
      (2) Crew Oxygen System
      (3) Passenger Oxygen System
      (4) Therapeutic Oxygen Outlets
      (5) Portable Oxygen Smoke Set
      (6) Limitations
    b. Controls and Components
      (1) Check mask and regulator
      (2) Crew and Passenger Oxygen System
      (3) Passenger supply
    c. Indicators/Indications
      (1) Oxygen
    d. Servicing/Preflight
      (1) Check Oxygen Quantity
      (2) Portable Oxygen Bottle
      (3) Smoke Goggles
      (4) Two Eros oxygen masks
  2. Operational Elements
    a. Check quantity and operation of the oxygen system
    b. Turn master supply valve on
  3. Limitation Elements
    a. Oxygen Type
    b. Appropriate FARs
  4. Emergency/Abnormal Procedure Elements
    a. Fuselage Fire
    b. Smoke Drill
    c. Annunciators from 1. R. I. d.

S. Aircraft Performance Module
  1. Elements
    a. Definitions
    b. Takeoff Weight to Meet Aircraft Certification Takeoff and Climb Criteria
      (1) Takeoff Thrust Setting
      (2) Maximum Continuous Thrust Setting
      (3) Maximum Takeoff Weight
      (4) Takeoff Speeds and Field Length
      (5) 2nd Segment Gradient
      (6) 2nd, 3rd and 4th Segment Gradient
      (7) Takeoff Flight Path Profile and Charts
      (8) Maximum Landing Weight
      (9) TODA vs. ASDA
      (10) TORA vs. ASDA
      (11) Max Takeoff Weight for Value of “D”
      (12) Approach Climb Gradient
      (13) Balked Landing Climb Gradient
      (14) Landing Distance
      (15) Turn Around Time
      (16) Practice Problems
      (17) Tabulated Data
T. Weight and Balance Module
  1. Elements
    a. Terminology
    b. Basic Empty Weight and Balance Record
    c. Weight and Balance Loading Form
    d. Cabin Arrangement Diagram
    e. Zero Fuel Weight
    f. Moments Limits vs. Weight
    g. Limitation Elements
    h. Practice Problems

U. Flight Planning Module
   1. Elements
      a. Rules of Thumb
      b. Reserve Fuel
      c. Section Charts
      d. Diversion Charts
      e. Other Tabulated Data

SYSTEMS INTEGRATION

TRAINING HOURS:

	HOURS
Aircraft Preflight Walkaround	1.5
High Altitude Training (if required)	1.5
Cockpit Procedures Mockup	1.5
Crew Resource Management	A/R

I. Systems Integration Module
   A. Aircraft Preflight Walkaround
   B. High Altitude Training Module-As required [IAW FAR 61.31 (f)(1)(i)]
      1. General Elements
         a. Respiratory System
         b. Circulation System
         c. Hypoxia; Effects, Causes, Symptoms
         d. Other High Altitude Sicknesses
         e. Causes and Effects of Gas Bubble Formation
         f. Causes and Effects of Gas Expansion
         g. Preventative Measures for Eliminating Gas Bubble Expansion
         h. Physiological Aspects of High Altitude Flight
         i. Physical Phenomena and Incidents of Decompression
      2. Operational Elements
         a. High Altitude Aerodynamics
         b. High Altitude Meteorology
         c. Normal Cruise Operations above 25000 Feet MSL

3. Limitations
    a. Duration of Consciousness Without Supplemental Oxygen
    b. Effects of Prolonged Use of Supplemental Oxygen
4. Emergency Procedures
    a. Emergency Procedures for a Rapid Decompression
    b. Emergency Descent Procedures

C. Cockpit Procedures Mockup
    1. The CPM will be used during the simulator prebriefs to discuss the aircraft systems that will be involved during that simulator.

CREW RESOURCE MANAGEMENT

CRM is briefed, monitored and practiced in each Simulator/Aircraft Session. The subjects taught are:

- Situational Awareness and the Error Chain
- Communication
- Synergy and Crew Concept
- Workload Assessment and Time Management
- Briefing
- Reliance on Automation
- Decision Making and Judgment
- Stress

FLIGHT TRAINING

TRAINING HOURS

Each flight simulator period is approximately 2.0 hours in length and incorporates an additional 1.0 hour for prebriefing and 0.5 hour for debriefing. Prebrief and postbrief times are reflected in the Ground School Training Hours Summary section.

Training is generally conducted as a crew; however, a pilot training alone may complete the course. The training hours for pilots training as a crew are specified in the table on the next page. The training hours for a pilot not training as a crew are specified in the table on the following page.

FLIGHT HOURS

INITIAL TRAINING As a Crew	Left Seat (Hours)	Right Seat (Hours)	Total Time (Hours)
Simulator Period No. 1	2.0	2.0	4.0
Simulator Period No. 2	2.0	2.0	4.0
Simulator Period No. 3	2.0	2.0	4.0
Simulator Period No. 4	2.0	2.0	4.0
Simulator Period No. 5	2.0	2.0	4.0
Simulator Period No. 6	2.0	2.0	4.0
LOFT	2.0	2.0	4.0
Aircraft Flight Training[1]			
TOTAL	14.0	14.0	28.0

[1] May vary as necessary

TRANSITION TRAINING As a Crew	Left Seat (Hours)	Right Seat (Hours)	Total Time (Hours)
Flight Simulator Period No. 1	2.0	2.0	4.0
Flight Simulator Period No. 2	2.0	2.0	4.0
Flight Simulator Period No. 3	2.0	2.0	4.0
Flight Simulator Period No. 4	2.0	2.0	4.0
Flight Simulator Period No. 5	2.0	2.0	4.0
Flight Simulator Period No. 6	2.0	2.0	4.0
LOFT	2.0	2.0	4.0
Aircraft Flight Training[2]			
TOTAL	14.0	14.0	28.0

[2] If required

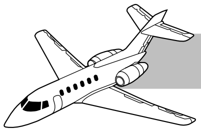
INITIAL TRAINING Not as a Crew	Left Seat (Hours)	Right Seat (Hours)	Total Time (Hours)
Flight Simulator Period No. 1	2.0	0.0	2.0
Flight Simulator Period No.2	2.0	0.0	2.0
Flight Simulator Period No. 3	2.0	0.0	2.0
Flight Simulator Period No. 4	2.0	0.0	2.0
Flight Simulator Period No. 5	2.0	0.0	2.0
Flight Simulator Period No. 6	2.0	0.0	2.0
LOFT	4.0	0.0	4.0
Aircraft Flight Training[1]			
TOTAL	16.01	0.0	16.0

[1] If required

TRANSITION TRAINING Not as a Crew	Left Seat (Hours)	Right Seat (Hours)	Total Time (Hours)
Flight Simulator Period No. 1	2.0	0.0	2.0
Flight Simulator Period No. 2	2.0	0.0	2.0
Flight Simulator Period No. 3	2.0	0.0	2.0
Flight Simulator Period No. 4	2.0	0.0	2.0
Flight Simulator Period No. 5	2.0	0.0	2.0
Flight Simulator Period No. 6	2.0	0.0	2.0
LOFT	4.0	0.0	4.0
Aircraft Flight Training[2]			
TOTAL	16.0	0.0	16.0

[2] If required

#Either item 9 or items 10, 11, 12



FLIGHT TRAINING MODULE OUTLINES

1. Flight Training Curriculum Segment

A. Objective: With the use of an approved flight simulator, cockpit checklist, and appropriate instrument approach and airport charts, the pilot will be able to accomplish the Normal and Emergency/Abnormal checklists, perform selected maneuvers and procedures, and implement Cockpit Resource Management techniques.

B. Training Equipment and Location

1. Simulators—No. 8347, No.8288, No. 8046 and No. 8263.
2. Cockpit Poster Panel—Briefing Room
3. Cockpit Checklists—Simulator and Briefing Room
4. Instrument Approach and Airport Charts—Simulator and Briefing Room

C. Maneuvers and Procedures Document (See Appendix A)

NOTE

The asterisk (*) symbol, in the following outlines, indicates an optional training item; that is, the instructor has the option to review this item as time permits. The identified items are also noted in Chapter 2. Items identified by the (¡) symbol are mandatory and the instructor will select the appropriate simulator session.

NOTE

Items from a numbered simulator module may be accomplished on a previous or subsequent module provided appropriate documentation is in the remarks section of the training record.

D. Simulator Period No. 1 Module

  1. Normal Procedure Events

    a. Normal Procedure Checklists

      (1) Preliminary Cockpit
      (2) Before Start-External, APU or Battery
      (3) Starting Engines-Extemal, APU or Battery
      (4) Failure to Start
      (5) After Start
      (6) Taxi
      (7) Before Takeoff
      (8) Line Up
      (9) Immediate Return/Traffic Pattern Checklist
      (10) After Takeoff and Climb

- (11) Descent and In Range
- (12) Before Landing
- (13) Close In
- (14) Missed Approach
- (15) After Landing
- (16) Shutdown
- (17) Postflight

b. Functional Checks

- (1) EFIS Ground Test—Collins
- (2) Air Data Computer Check—Collins
- (3) Individual Instrument Test—Collins
- (4) Pre-Flight Check-Autopilo
- (5) Electric Pitch Trim
- (6) Stall Systems Test
- (7) EFIS Self Test—Sperry
- (8) Air Data Computer Test—Honeywell Phase III
- (9) Thrust Reverser Check
- (10) Computer and Rudder Bias Check
- (11) APR Check
- (12) Brake Checks-Taxi, Before Landing
- (13) Periodical Systems Check
- (14) Solar APU/Garrett APU

2. Abnormal/Emergency Procedures

a. Engine/APU/Fuel

- (1) Engine Fire Warning on the Ground
- (2) * APU Fire
- (3) * APU Failure

b. Flight Controls

- (1) Stick Shaker and/or Pusher Operation
- (2) Stall Ident Warnings

- c. Avionics/EFIS
    - (1) * Display Processor Failure—Collins
    - (2) * Display Control Failure—Collins
    - (3) * EADI Failure—Collins
    - (4) * EHSI Failure—Collins
    - (5) * EFIS Fan Failure—Collins
    - (6) * EFIS Fan Failure—Sperry
    - (7) * EFIS Fan Fail Lts Illuminated Inflight—Sperry
    - (8) * Electronic Display Indicator Over-temperature—Sperry
    - (9) * EFIS Display Failure
    - (10) * Flight Guidance System—Sperry
    - (11) * AHRS and O'Heat Lights On—Sperry
    - (12) * Air Data Computer Failure—Sperry

3. Flight Training Events
    - a. Preparation
        - (1) Prestart Procedures
    - b. Surface Operation
        - (1) Starting
        - (2) Taxi
        - (3) Pretakeoff Checks
    - c. Takeoff
        - (1) Normal Takeoff
        - (2) Crosswind Takeoff
        - (3) Instrument Takeoff
        - (4) Area Departure
    - d. Climb
        - (1) Normal
    - e. Descent
        - (1) Normal
    - f. Approaches
        - (1) Area Arrival
        - (2) Navigation Equipment and Assigned Radials
        - (3) Precision ILS
        - (4) Nonprecision FMS Approach

g. Landings
(1) Normal Landing
h. After Landing
(1) Parking
i. System Procedures ~(Normal, Abnormal, Alternate)
(1) Autopilot
(2) Flight Management Guidance Systems and/or Automatic or other Approach and Landing Systems
(3) Communications Equipment
(4) Navigation Systems
4. Crew Resource Management
a. * Situational Awareness and the Error Chain
b. * Communication
c. * Synergy and Crew Concept
d. * Workload Assessment and Time Management
e. * Briefing
* Reliance on Automation
g. * Decision Making and Judgement
h. * Stress
E. Simulator Period No. 2 Module
1. Normal Procedure Events
a. Normal Procedure Checklists
(1) Preliminary Cockpit
(2) Before Start-External APU, or Battery
(3) Starting Engines-External, APU, or Battery
(4) After Start
(5) Taxi
(6) Before Takeoff
(7) Line Up
(8) Immediate Return/Traffic Pattern Checklist
(9) After Takeoff and Climb
(10) Cruise
(11) Descent and In Range
(12) Before Landing

(13) Close In
(14) Missed Approach
(15) After Landing
(16) Shutdown
(17) Post Flight

b. Functional Checks
(1) EFIS Ground Test—Collins
(2) Air Data Computer Check—Collins
(3) Individual Instrument Test—Collins
(4) Pre-Flight Check-Autopilot and Flight Guidance System (if applicable)
(5) Electric Pitch Trim
(6) Stall System Test
(7) EFIS Self Test—Sperry
(8) Air Data Computer Test—Honeywell Phase III
(9) Thrust Reverser Check
(10) Computer and Rudder Bias Check
(11) APR Check
(12) Brake Checks-Taxi, Before Landing
(13) Periodical Systems Check
(14) Solar APU/Garrett APU

2. Abnormal/Emergency Procedures

a. Electrical
(1) Double Generator Failure
(2) * Single Generator Failure
(3) * Abnormal Voltage
(4) * Voltage Fluctuation
(5) * Generator Out of Balance
(6) Battery Overheat
(7) Battery Hot
(8) Battery Contactor
(9) Double Inverter Failure
(10) XS1 or XS2 Fail
(11) Single Inverter Failure
(12) XE Fail

- (13) XS I Fail and XE Fail
- (14) Emergency Battery Supply
- (15) * Smoke Drill

b. Engine/APU/Fuel

- (1) Engine Failure
- (2) Engine Relight
- (3) Immediate Relight
- (4) Relight Envelope
- (5) Starter Assisted (Normal) Relight
- (6) Windmill Relight
- (7) * Engine Compressor Stall
- (8) * Engine Computer Malfunction
- (9) Abnormal Indications or Behavior
- (10) * Oil Low Pressure
- (11) * APU Fire
- (12) * APU Failure
- (13) * Engine Fuel Malfunction
- (14) Fuel Low Pressure
- (15) * Auxiliary Fuel Transfer Failure
- (16) Fuel Management After One Engine Failure

c. Avionics/EFIS

- (1) * Display Processor Failure-Collins
- (2) * Display Control Failure—Collins
- (3) * EADI Failure—Collins
- (4) * EHSI Failure—Collins
- (5) * EFIS Fan Failure—Collins
- (6) * EFIS Fan Failure—Sperry
- (7) * EFIS Fan Fail Lts Illuminated Inflight—Sperry
- (8) * Electronic Display Indicator Over-temperature—Sperry
  - (9) * EFIS Display Failure
  - (10) * Flight Guidance System—Sperry
  - (11) * AHRS and O'Heat Lights On—Sperry
- (12) * Air Data Computer Failure—Sperry

- d. Ice and Rain Protection
    - (1) Windscreen Overheat
    - (2) Sidescreen Overheat
    - (3) * Alternator Failure
    - (4) * Double Alternator Failure
    - (5) * Air Data Computer Failure

3. Flight Training Events
    - a. Preparation
        - (1) Prestart Procedures
    - b. Surface Operation
        - (1) Starting
        - (2) Taxi
        - (3) Pretakeoff Checks
    - c. Takeoff
        - (1) Normal Takeoff
        - (2) Crosswind Takeoff
        - (3) Instrument Takeoff
    - d. Climb
        - (1) Normal
    - e. En Route
        - (1) Steep Turns
        - (2) Approach to Stall—Clean Configuration
        - (3) Approach to Stall—Takeoff Flaps
        - (4) Approach to Stall—Landing Configuration
        - (5) Unusual Attitudes [Part 61]
        - (6) Inflight Powerplant Shutdown
        - (7) Inflight Powerplant Restart
    - f. Descent
        - (1) Normal

g. Approaches
  (1) Area Departure and Arrival
  (2) Holding
  (3) Precision ILS
  (4) * Nonprecision—VOR
  (5) * Nonprecision—NDB
  (6) Missed Approach from a Precision Approach
  (7) Missed Approach from a Nonprecision Approach
h. Landings
  (1) Normal Landing
  (2) Landing from a Precision Instrument Approach
  (3) Crosswind Landing
i. After Landing
  (1) Parking
j. Other Flight Procedures
  (1) Holding
k. System Procedures (Normal, Abnormal, Altemate)
  (1) Fuel and Oil
  (2) Electrical
  (3) Autopilot
  (4) Flight Management Guidance Systems and/or Automatic or other Approach and Landing Systems
  (5) Stall Warning Devices
  (6) Flight Instrument System Malfunction
  (7) Communications Equipment
  (8) Navigation Systems
l. System Procedures (Emergency)
  (1) Smoke Control
  (2) Electrical Systems

4. Crew Resource Management
   - a. *Situational Awareness and the Error Chain
   - b. * Communication
   - c. * Synergy and Crew Concept
   - d. * Workload Assessment and Time Management
   - e. * Briefing
   - f. * Reliance on Automation
   - g. * Decision Making and Judgement
   - h. * Stress

F. Simulator Period No. 3 Module

1. Normal Procedure Events
   - a. Normal Procedure Checklists
     - (1) * Preliminary Cockpit
     - (2) Thru Flight
     - (3) Before Start—External, APU or Battery
     - (4) * Starting Engines-Extemal, APU or Battery
     - (5) After Start
     - (6) Taxi
     - (7) Before Takeoff
     - (8) Line Up
     - (9) Immediate Return/Traffic Pattern Checklist
     - (10) After Takeoff and Climb
     - (11) Cruise
     - (12) Descent and In Range
     - (13) Before Landing
     - (14) Close In
     - (15) Missed Approach
     - (16) After Landing
     - (17) Shutdown
     - (18) Postflight

b. Functional Checks
    (1) EFIS Ground Test—Collins
    (2) Air Data Computer Check—Collins
    (3) Individual Instrument Test—Collins
    (4) Pre-Flight Check-Autopilot and Flight Guidance System (if applicable)
    (5) Electric Pitch Trim
    (6) Stall Systems Test
    (7) EFIS Self Test-Sperry
    (8) Air Data Computer Test—Honeywell Phase III
    (9) Thrust Reverser Check
    (10) Computer and Rudder Bias Check
    (11) APR Check
    (12) Brake Checks—Taxi, Before Landing
    (13) Solar APU/Garrett APU

2. Abnormal/Emergency Procedures
    a. Engine/APU/Fuel
        (1) Engine Fire Warning Inflight
        (2) Engine Failure
        (3) Engine Relight
        (4) Immediate Relight
        (5) Relight Envelope
        (6) Starter Assisted (Normal) Relight
        (7) * Windmill Relight
        (8) * Engine Compressor Stall
        (9) * Engine Computer Malfunction
        (10) * Abnormal Indications or Behavior
        (11) * Oil Low Pressure
        (12) * APU Fire
        (13) * APU Failure
        (14) * Engine Fuel Malfunction
        (15) * Fuel Low Pressure
        (16) * Auxiliary Fuel Transfer Failure
        (17) * Fuel Management After One Engine Failure

b. Hydraulics/Gear/Brakes
  (1) Hydraulic Low Pressure
  (2) Main Hydraulic Failure
  (3) * Hydraulic System Overheat
  (4) Normal Brake Supply Failure
  (5) * Emergency Brake Supply Failure
  (6) * Failure of Landing Gear to Retract
  (7) * Failure of Landing Gear Position Indicator
  (8) * Landing with Gear in Abnormal Positions

c. Thrust Reversers
  (1) Thrust Reverser Malfunctions In Flight

d. Avionics/EFIS
  (1) * Display Processor Failure—Collins
  (2) * Display Control Failure—Collins
  (3) * EADI Failure—Collins
  (4) * EHSI Failure—Collins
  (5) * EFIS Fan Failure—Collins
  (6) * EFIS Fan Failure—Sperry
  (7) * EFIS Fan Fail Lts Illuminated Inflight—Sperry
  (8) * Electronic Display Indicator Over-temperature—Sperry
  (9) * EFIS Display Failure
  (10) * Flight Guidance System—Sperry
  (11) * AHRS and O'Heat Lights On—Sperry
  (12) * Air Data Computer Failure—Sperry

e. Ice and Rain Protection
  (1) * Windscreen Overheat
  (2) * Sidescreen Overheat
  (3) * Alternator Failure
  (4) * Double Alternator Failure
  (5) * Air Data Computer Failure

3. Flight Training Events (all events may be combined as necessary)

a. Preparation
  (1) * Prestart Procedures

b. Surface Operation
    (1) Starting
    (2) Taxi
    (3) Pretakeoff Checks

c. Takeoff
    (1) Normal Takeoff
    (2) Crosswind Takeoff
    (3) Rejected (Aborted) Takeoff
    (4) Instrument Takeoff
    (5) Power Failure V_1 (at or above V_1)

d. Climb
    (1) Normal

e. En Route
    (1) Inflight Powerplant Shutdown
    (2) * Inflight Powerplant Restart

f. Descent
    (1) Normal

g. Approaches
    (1) Area Departure and Arrival
    (2) Holding
    (3) Precision ILS
    (4) Precision ILS with an Engine Out
    (5) Nonprecision VOR
    (6) Nonprecision—NDB
    (7) Nonprecision with an Engine Out
    (8) Circling Approach
    (9) Missed Approach from a Precision Approach
    (10) Missed Approach from a Nonprecision Approach
    (11) Missed Approach with a Powerplant Failure

h. Landings
    (1) Normal Landing
    (2) Landing from a Precision Instrument Approach
    (3) Landing from a Precision Instrument Approach—One Engine Inop
    (4) Landing with a Flap Malfunction
    (5) Crosswind Landing
    (6) Landing from a Circling Approach
    (7) Rejected Landing to a Normal Missed Approach

i. After Landing
    (1) Parking

j. Other Flight Procedures
    (1) Holding
    (2) Air Hazard Avoidance
    (3) Windshear/Microburst

k. System Procedures (Normal, Abnormal, Altemate)
    (1) Fuel and Oil
    (2) * Electrical
    (3) Hydraulic
    (4) Autopilot
    (5) Flight Management Guidance Systems and/or Automatic or other Approach and Landing Systems
    (6) Airborne Weather Radar
    (7) Flight Instrument System Malfunction
    (8) Communications Equipment
    (9) Navigation Systems

l. System Procedures (Emergency)
    (1) Aircraft Fires
    (2) * Smoke Control
    (3) Powerplant Malfunctions
    (4) Electrical Systems
    (5) Hydraulic Systems
    (6) * Flight Control Systems Malfunction
    (7) Landing Gear and Flap System Malfunction

4. Crew Resource Management
   a. * Situational Awareness and the Error Chain
   b. * Communication
   c. * Synergy and Crew Concept
   d. * Workload Assessment and Time Management
   e. * Briefing
   f. * Reliance on Automation
   g. * Decision Making and Judgement
   h. * Stress

G. Simulator Period No. 4 Module
1. Normal Procedure Events
   a. Normal Procedure Checklists
      (1) * Preliminary Cockpit
      (2) * Thru Flight
      (3) * Before Start—Extemal, APU, or Battery
      (4) * Starting Engines-External, APU, or Battery
      (5) * After Start
      (6) Taxi
      (7) Before Takeoff
      (8) Line Up
      (9) Immediate Return/Traffic Pattern Checklist
      (10) * After Takeoff and Climb
      (11) * Cruise
      (12) * Descent and In Range
      (13) Before Landing
      (14) Close In
      (15) Missed Approach
      (16) After Landing
      (17) * Shutdown
      (18) * Post Flight

b. Functional Checks (if needed for proficiency)
   (1) EFIS Ground Test-Collins
   (2) Air Data Computer Check—Collins
   (3) Individual Instrument Test—Collins
   (4) Pre-Flight Check-Autopilot and Flight Guidance System (if applicable)
   (5) Electric Pitch Trim
   (6) Stall System Test
   (7) EFIS Self Test—Sperry
   (8) Air Data Computer Test—Honeywell Phase III
   (9) Thrust Reverser Check
   (10) Computer and Rudder Bias Check
   (11) APR Check
   (12) Brake Checks—Taxi, Before Landing
   (13) Solar APU

2. Abnormal/Emergency Procedures
   a. Engine/APU/Fuel
      (1) Engine Failure
      (2) * Engine Relight
      (3) * Immediate Relight
      (4) * Relight Envelope
      (5) * Starter Assisted (Normal) Relight
      (6) * Windmill Relight
      (7) * Engine Compressor Stall
      (8) * Engine Computer Malfunction
      (9) * Abnormal Indications or Behavior
      (10) * Oil Low Pressure
      (11) * APU Fire
      (12) * APU Failure
      (13) * Engine Fuel Malfunction
      (14) Fuel Low Pressure
      (15) * Auxiliary Fuel Transfer Failure

b. Pressurization/Pneumatics
    (1) Depressurization or Complete Loss of Air Supply
    (2) Emergency Descent
    (3) Failure to Pressurization
    (4) Automatic Pressure Control Failure
    (5) * Duct Overheat
    (6) * Automatic Temperature Control Failure
    (7) Rear Equipment Bay Overheat
    (8) High Pressure Air Overheat

c. Thrust Reversers
    (1) * Thrust Reverser Malfunctions in Flight

d. Avionics/EFIS
    (1) * Display Processor Failure—Collins
    (2) * Display Control Failure—Collins
    (3) * EADI Failure—Collins
    (4) * EHSI Failure—Collins
    (5) * EFIS Fan Failure—Collins
    (6) * EFIS Fan Failure—Sperry
    (7) * EFIS Fan Fail Lts Illuminated Inflight—Sperry
    (8) * Electronic Display Indicator Over-temperature—Sperry
    (9) * EFIS Display Failure
    (10) * Flight Guidance System—Speffy
    (11) * AHRS and O'Heat Lights On—Sperry
    (12) * Air Data Computer Failure—Sperry

e. Ice and Rain Protection
    (1) Windscreen Overheat
    (2) Sidescreen Overheat
    (3) Alternator Failure
    (4) Double Alternator Failure
    (5) Air Data Computer Failure
    (6) Engine and Airframe Anti-ice

3. Flight Training Events (events may be combined as necessary)

a. Preparation
    (1) * Prestart Procedures

b. Surface Operation
    (1) Starting
    (2) Taxi
    (3) Pretakeoff Checks
c. Takeoff
    (1) Normal Takeoff
    (2) Normal Noise Abatement Takeoff
    (3) Normal Takeoff (in ice, and use of Engine Anti-ice and TKS)
    (4) Crosswind Takeoff
    (5) Rejected (Aborted) Takeoff
    (6) Instrument Takeoff
    (7) Power Failure V_1 (at or above V_1)
d. Climb
    (1) Normal
e. En Route
    (1) Inflight Powerplant Shutdown
    (2) * Inflight Powerplant Restart
    (3) High Speed Handling
f. Descent
    (1) Normal
    (2) Maximum Rate Descent
g. Approaches
    (1) Area Departure and Arrival
    (2) * Holding
    (3) Visual Approach with a Flap Malftunction
    (4) Precision ILS
    (5) * Precision ILS—with an Engine Out
    (6) * Nonprecision—VOR
    (7) * Nonprecision—NDB
    (8) * Nonprecision with an Engine Out
    (9) Circling Approach
    (10) Missed Approach from a Precision Approach
    (11) * Missed Approach from a Nonprecision Approach
    (12) Missed Approach with a Powerplant Failure

h. Landings
    (1) * Normal Landing
    (2) Landing from a Precision Instrument Approach
    (3) Landing from a Precision Instrument Approach—One Engine Inop
    (4) Landing with a Flap Malfunction
    (5) * Crosswind Landing
    (6) Maneuver to Landing with a Powerplant Failure
    (7) * Landing from a Circling Approach
    (8) Rejected Landing to a Normal Missed Approach
i. After Landing
    (1) Parking
j. Other Flight Procedures
    (1) Holding
    (2) Ice Accumulation on Airframe
    (3) Air Hazard Avoidance
k. System Procedures (Normal, Abnormal, Alternate)
    (1) Pneumatic/Pressurization
    (2) Air Conditioning
    (3) * Fuel and Oil
    (4) * Electrical
    (5) * Hydraulic
    (6) Flight Controls
    (7) Anti-icing and Deicing
    (8) * Autopilot
    (9) Flight Management Guidance Systems and/or Automatic or other Approach and Landing Systems
    (10) Airborne Weather Radar
    (11) Flight Instrument System Malfunction
    (12) Communications Equipment
    (13) Navigation Systems

l. System Procedures (Emergency)
  - (1) * Aircraft Fires
  - (2) * Smoke Control
  - (3) Powerplant Malfunctions
  - (4) * Electrical Systems
  - (5) * Hydraulic Systems
  - (6) * Pneumatic Systems
  - (7) * Flight Control Systems Malfunction
  - (8) * Landing Gear and Flap System Malfunction

4. Crew Resource Management
  - a. * Situational Awareness and the Error Chain
  - b. * Communication
  - c. * Synergy and Crew Concept
  - d. * Workload Assessment and Time Management
  - e. * Briefing
  - f. * Reliance on Automation
  - g. * Decision Making and Judgment
  - h. * Stress

H. Simulator Period No. 5 Module

1. Normal Procedure Events
  - a. Normal Procedure Checklists
    - (1) * Preliminary Cockpit
    - (2) * Before Start-External,APU, or Battery
    - (3) * Starting Engines—External, APU, or Battery
    - (4) * After Start
    - (5) Taxi
    - (6) Before Takeoff
    - (7) Line Up
    - (8) Immediate Return/Traffic Pattern Checklist
    - (9) * After Takeoff and Climb
    - (10) * Cruise
    - (11) * Descent and In Range
    - (12) Before Landing
    - (13) Close In

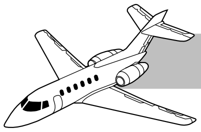
(14) Missed Approach
(15) After Landing
(16) * Shutdown
(17) * Post Flight

b. Functional Checks
(1) * EFIS Ground Test—Collins
(2) * Air Data Computer Check—Collins
(3) * Individual Instrument Test—Collins
(4) * Pre-Flight Check-Autopilot and Flight Guidance System (if applicable)
(5) * Electric Pitch Trim
(6) * Stall System Test
(7) * EFIS Self Test—Speffy
(8) * Air Data Computer Test—Honeywell Phase III
(9) * Thrust Reverser Check
(10) * Computer and Rudder Bias Check
(11) * APR Check
(12) * Brake Checks-Taxi, Before Landing
(13) * Solar APU/Garrett APU

2. Abnormal/Emergency Procedures

a. Electrical
(1) * Double Generator Failure
(2) * Single Generator Failure
(3) * Abnormal Voltage
(4) * Voltage Fluctuation
(5) * Generator Out of Balance
(6) * Battery Overheat
(7) * Battery Hot
(8) Battery ~Contactor
(9) Double Inverter Failure
(10) XS1 or XS 2 Fail
(11) Single Inverter Failure
(12) XE Fail
(13) XS1 Fail and XE Fail
(14) Emergency Battery Supply
(15) Smoke Drill

b. Engine/APU/Fuel
  (1) * Engine Fire Warning Light
  (2) * Engine Fire Warning on the Ground
  (3) * Engine Failure
  (4) * Engine Relight
  (5) * Immediate Relight
  (6) * Relight Envelope
  (7) * Starter Assisted (Normal) Relight
  (8) * Windmill Relight
  (9) * Engine Compressor Stall
  (10) * Engine Computer Malfunction
  (11) * Abnormal Indications or Behavior
  (12) * Oil Low Pressure
  (13) * APU Fire
  (14) * APU Failure
  (15) * Engine Fuel Malfunction
  (16) * Fuel Low Pressure
  (17) * Auxiliary Fuel Transfer Failure
  (18) * Fuel Management After One Engine Failure
c. Pressurization/Pneumatics
  (1) * Depressurization or Complete Loss of Air Supply
  (2) * Emergency Descent
  (3) * Failure to Pressurize
  (4) * Automatic Pressure Control Failure
  (5) * Duct Overheat
  (6) * Automatic Temperature Control Failure
  (7) * Rear Equipment Bay Overheat
  (8) * High Pressure Air Overheat



d. Hydraulics/Gear/Brakes
(1) * Hydraulic Low Pressure
(2) * Main Hydraulic Failure
(3) * Hydraulic System Overheat
(4) * Normal Brake Supply Failure
(5) * Emergency Brake Supply Failure
(6) * Failure of Landing Gear to Retract
(7) * Nosewheel not Centered
(8) * Landing Gear Three Greens not Indicated

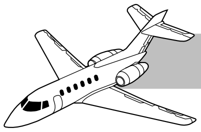
e. Flight Controls
(1) Elevator Trim Runaway

f. Thrust Reverser
(1) Thrust Reverser Malfunctions In Flight

g. Avionics/EFIS
(1) * Display Processor Failure—Collins
(2) * Display Control Failure—Collins
(3) * EADI Failure—Collins
(4) * EHSI Failure—Collins
(5) * EFIS Fan Failure—Collins
(6) * EFIS Fan Failure—Sperry
(7) * EFIS Fan Fail Lts Illuminated Inflight—Sperry
(8) * Electronic Display Indicator Over-temperature—Sperry
(9) * EFIS Display Failure
(10) * Flight Guidance System—Sperry
(11) * AHRS and O'Heat Lights On—Sperry
(12) * Air Data Computer Failure—Sperry

h. Ice and Rain Protection
(1) * Windscreen Overheat
(2) * Sidescreen Overheat
(3) * Alternator Failure
(4) * Double Alternator Failure
(5) * Air Data Computer Failure

3. Flight Training Events (events may be combined as necessary)
   - a. Preparation
     - (1) * Prestart Procedures
   - b. Surface Operation
     - (1) * Starting
     - (2) * Taxi
     - (3) * Pretakeoff Checks
   - c. Takeoff
     - (1) Normal Takeoff
     - (2) Crosswind Takeoff
     - (3) Rejected (Aborted) Takeoff
     - (4) Instrument Takeoff
     - (5) Power Failure V_1 (at or above V_1)
     - (6) Lower than Standard Minimum Takeoff (FAR 135 clients only)
   - d. Climb
     - (1) Normal
   - e. En Route
     - (1) Inflight Powerplant Shutdown
     - (2) * Inflight Powerplant Restart
   - f. Descent
     - (1) Normal
     - (2) * Maximum Rate Descent (Emergency Descent)
   - g. Approaches
     - (1) Area Departure and Arrival
     - (2) Navigation Equipment and Assigned Radials
     - (3) * Holding
     - (4) * Precision ILS
     - (5) Precision ILS with an Engine Out
     - (6) * Nonprecision—VOR
     - (7) * Nonprecision—NDB
     - (8) Nonprecision with an Engine Out
     - (9) * Circling Approach
     - (10) * Missed Approach from a Precision Approach
     - (11) * Missed Approach from a Nonprecision Approach
     - (12) * Missed Approach with a Powerplant Failure



h. Landings
    (1) * Normal Landing
    (2) Landing from a Precision Instrument Approach
    (3) * Crosswind Landing
    (4) * Maneuver to Landing with a Powerplant Failure from a Visual Approach
    (5) * Landing from a Circling Approach
    (6) * Rejected Landing to a Normal Missed Approach
i. After Landing
    (1) Parking
j. Other Flight Procedures
    (1) * Holding
    (2) * Ice Accumulation on Airframe
    (3) Air Hazard Avoidance
    (4) Windshear/Microburst
k. System Procedures ~(Normal, Abnormal, Alternate)
    (1) * Pneumatic/Pressurization
    (2) * Air Conditioning
    (3) * Fuel and Oil
    (4) * Electrical
    (5) * Hydraulic
    (6) * Flight Controls
    (7) * Anti-icing and Deicing
    (8) * Autopilot
    (9) Flight Management Guidance Systems and/or Automatic or other Approach and Landing Systems
    (10) Airborne Weather Radar
    (11) * Flight Instrument System Malfunction
    (12) Communications Equipment
    (13) Navigation Systems

- l. System Procedures (Emergency)
    - (1) * Aircraft Fires
    - (2) * Smoke Control
    - (3) Powerplant Malfunctions
    - (4) * Electrical Systems
    - (5) * Hydraulic Systems
    - (6) * Pneumatic Systems
    - (7) * Flight Control Systems Malfunction
    - (8) * Landing Gear and Flap System Malfunction
- 4. Crew Resource Management
    - a. * Situational Awareness and the Error Chain
    - b. * Communication
    - c. * Synergy and Crew Concept
    - d. * Workload Assessment and Time Management
    - e. * Briefing
    - f. * Reliance on Automation
    - g. * Decision Making and Judgment
    - h. * Stress

- I. Simulator Period No. 6 Module
    - 1. Normal Procedure Events
        - a. Normal Procedure Checklists
            - (1) * Preliminary Cockpit
            - (2) Before Start—Extemal, APU, or Battery
            - (3) * Starting Engines—External, APU, or Battery
            - (4) After Start
            - (5) Taxi
            - (6) Before Takeoff
            - (7) Line Up
            - (8) Immediate Return/Traffic Pattern Checklist
            - (9) After Takeoff and Climb
            - (10) Cruise
            - (11) Descent and In Range
            - (12) Before Landing
            - (13) Close In

    - (14) Missed Approach
    - (15) After Landing
    - (16) Shutdown
    - (17) Post Flight
  - b. Functional Checks
    - (1) EFIS Ground Test—Collins
    - (2) Air Data Computer Check—Collins
    - (3) Individual Instrument Test—Collins
    - (4) Pre-Flight Check—Autopilot and Flight Guidance System (if applicable)
    - (5) Electric Pitch Trim
    - (6) Stall System Test
    - (7) EFIS Self Test—Sperry
    - (8) Air Data Computer Test—Honeywell Phase III
    - (9) Thrust Reverser Check
    - (10) Computer and Rudder Bias Check
    - (11) APR Check
    - (12) Brake Checks-Taxi, Before Landing
    - (13) Solar APU
- 2. Abnormal/Emergency Procedures
  - a. Electrical
    - (1) * Double Generator Failure
    - (2) * Single Generator Failure
    - (3) * Abnormal Voltage
    - (4) * Voltage Fluctuation
    - (5) * Generator Out of Balance
    - (6) * Battery Overheat
    - (7) * Battery Hot
    - (8) Battery Contactor
    - (9) Double Inverter Failure
    - (10) XS1 or XS2 Fail
    - (11) Single Inverter Failure
    - (12) XE Fail
    - (13) XS1 Fail and XE Fail
    - (14) Emergency Battery Supply
    - (15) Smoke Drill

b. Engine/APU/Fuel
    (1) * Engine Fire Warning Light
    (2) * Engine Fire Warning on the Ground
    (3) * Engine Failure
    (4) * Engine Relight
    (5) * Immediate Relight
    (6) * Relight Envelope
    (7) * Starter Assisted (Normal) Relight
    (8) * Windmill Relight
    (9) * Engine Compressor Stall
    (10) * Engine Computer Malfunction
    (11) * Abnormal Indications or Behavior
    (12) * Oil Low Pressure
    (13) * APU Fire
    (14) * APU Failure
    (15) * Engine Fuel Malfunction
    (16) * Fuel Low Pressure
    (17) * Auxiliary Fuel Transfer Failure
    (18) * Fuel Management After One Engine Failure

c. Pressurization/Pneumatics
    (1) * Depressurization or Complete Loss of Air Supply
    (2) * Emergency Descent
    (3) * Failure to Pressurize
    (4) * Automatic Pressure Control Failure
    (5) * Duct Overheat
    (6) * Automatic Temperature Control Failure
    (7) * Rear Equipment Bay Overheat
    (8) * High Pressure Air Overheat

d. Hydraulics/Gear/Brakes
    (1) * Hydraulic Low Pressure
    (2) * Main Hydraulic Failure
    (3) * Hydraulic System Overheat
    (4) * Normal Brake Supply Failure
    (5) * Emergency Brake Supply Failure
    (6) * Failure of Landing Gear to Retract
    (7) * Nosewheel not Centered
    (8) * Landing Gear Three Greens not Indicated
e. Flight Controls
    (1) Stick Shaker and/or Pusher Operation
    (2) Stall ~Ident Warnings
    (3) Elevator Trim Runaway
f. Thrust Reverser
    (1) * Inadvertent T/R Deployment During Flight
    (2) * Thrust Reverser Malfimctions In Flight
    (3) * Landing with One Thrust Reverser Deployment
g. Avionics/EFIS
    (1) * Display Processor Failure—Collins
    (2) * Display Control Failure—Collins
    (3) * EADI Failure—Collins
    (4) * EHSI Failure—Collins
    (5) * EFIS Fan Failure—Collins
    (6) * EFIS Fan Failure—Sperry
    (7) * EFIS Fan Fail Lts Illuminated Inflight—Sperry
    (8) Electronic Display Indicator Over-temperature—Sperry
    (9) * EFIS Display Failure
    (10) * Flight Guidance System—Sperry
    (11) * AHRS and O’Heat Lights On—Sperry
    (12) * Air Data Computer Failure—Sperry

h. Ice and Rain Protection
    (1) * Windscreen Overheat
    (2) * Sidescreen Overheat
    (3) * Alternator Failure
    (4) * Double Alternator Failure
    (5) * Air Data Computer Failure

3. Flight Training Events
  a. Preparation
    (1) * Prestart Procedures
  b. Surface Operation
    (1) Starting
    (2) Taxi
    (3) Pretakeoff Checks
  c. Takeoff
    (1) Normal Takeoff
    (2) Crosswind Takeoff
    (3) Rejected (Aborted) Takeoff
    (4) Instrument Takeoff
    (5) Power Failure V_1 (at or above V_1)
    (6) Lower than Standard Minimum Takeoff (FAR 135 clients only)
  d. Climb
    (1) Normal
  e. En Route
    (1) Steep Turns
    (2) Approach to Stall—Clean Configuration
    (3) Approach to Stall—Takeoff Flaps
    (4) Approach to Stall—Landing Configuration
    (5) Unusual Attitudes
    (6) Inflight Powerplant Shutdown
    (7) * Inflight Powerplant Restart
  f. Descent
    (1) Normal
    (2) * Maximum Rate Descent

g. Approaches
    (1) Area Departure and Arrival
    (2) Navigation Equipment and Assigned Radials
    (3) * Holding
    (4) * Visual Approach—One Engine Inoperative
    (5) Precision ILS
    (6) Precision ILS with an Engine Out
    (7) Nonprecision—VOR
    (8) Nonprecision—NDB
    (9) * Nonprecision with an Engine Out
    (10) Circling Approach
    (11) Missed Approach from a Precision Approach
    (12) Missed Approach from a Nonprecision Approach
    (13) Missed Approach with a Powerplant Failure
h. Landings
    (1) * Normal Landing
    (2) Landing from a Precision Instrument Approach
    (3) Landing with a Flap Malfunction
    (4) * Crosswind Landing
    (5) Maneuver to Landing with a Powerplant Failure from a Visual Approach
    (6) Landing from a Circling Approach
    (7) Rejected Landing to a Normal Missed Approach
i. After Landing
    (1) Parking
j. Other Flight Procedures
    (1) Holding
    (2) Ice Accumulation on Airframe
    (3) Air Hazard Avoidance
    (4) Windshear/Microburst
k. System Procedures (Normal, Abnormal, Alternate)
    (1) Pneumatic/Pressurization
    (2) Air Conditioning
    (3) Fuel and Oil
    (4) Electrical

- (5) Hydraulic
- (6) Flight Controls
- (7) Anti-icing and Deicing
- (8) Autopilot
- (9) Flight Management Guidance Systems and/or Automatic or other Approach and Landing Systems
- (10) Stall Warning Devices
- (11) Airborne Weather Radar
- (12) Flight Instrument System Malfunction
- (13) Communications Equipment
- (14) Navigation Systems

1. System Procedures (Emergency)
    - (1) * Aircraft Fires
    - (2) * Smoke Control
    - (3) Powerplant Malfunctions
    - (4) Electrical Systems
    - (5) Hydraulic Systems
    - (6) Pneumatic Systems
    - (7) * Flight Control Systems Malfunction
    - (8) Landing Gear and Flap System Malfunction

4. Crew Resource Management
    - a. * Situational Awareness and the Error Chain
    - b. * Communication
    - c. * Synergy and Crew Concept
    - d. * Workload Assessment and Time Management
    - e. * Briefing
    - f. * Reliance on Automation
    - g. * Decision Making and Judgment
    - h. * Stress

J. Aircraft Period No. 1 Module

1. Normal Procedure Events
    - a. Normal Procedure Checklists
        - (1) Preliminary Cockpit
        - (2) Exterior Inspection

(3) Interior Cabin
(4) Before Start-External, APU, or Battery
(5) Starting Engines-External, APU, or Battery
(6) After Start
(7) Taxi
(8) Before Takeoff
(9) Line Up
(10) Immediate Return/Traffic Pattern Checklist
(11) After Takeoff and Climb
(12) Cruise
(13) Descent and In Range
(14) Before Landing
(15) Close In
(16) Missed Approach
(17) After Landing
(18) Shutdown
(19) Post Flight

b. Functional Checks
(1) EFIS Ground Test—Collins
(2) Air Data Computer Check—Collins
(3) Individual Instrument Test—Collins
(4) Pre-Flight Check-Autopilot and Flight Guidance System (if applicable)
(5) Electric Pitch Trim
(6) Stall System Test
(7) EFIS Self Test-Sperry
(8) Air Data Computer Test—Honeywell Phase III
(9) Thrust Reverser Check
(10) Computer and Rudder Bias Check
(11) APR Check
(12) Brake Checks—Taxi, Before Landing
(13) Garrett APU
(14) Solar APU
(15) Refueling

2. Abnormal/Emergency Procedures
    a. Engine/APU/Fuel
        (1) Engine Failure
3. Flight Training Events
    a. Preparation
        (1) Visual Inspection
    b. Takeoff
        (1) Normal Takeoff
        (2) Crosswind Takeoff
        (3) Instrument Takeoff
        (4) Power Failure V_1 (at or above V_1)
    c. Approaches
        (1) Precision ILS
        (2) * Nonprecision Approach—VOR
        (3) * Nonprecision Approach—NDB
        (4) * Circling Approach
    d. Landings
        (1) Normal
        (2) * Simulated Single Engine
        (3) Crosswind
        (4) * From a Circling Approach

K. Line Orientated Flight Training (Transition)
1. LOFT Period No. I
    a. LOFT—Flight Events
        (1) Route of Flight Events
            (a) Denver to Salt Lake City
            (b) White Plains to Washington
            (c) Houston to Atlanta
            (d) Atlanta to Washington
            (e) San Francisco to Los Angeles
            (f) Los Angeles to San Francisco
            (g) Boston to New York
            (h) New York to Boston

Note

Each LOFT will include any two (2) of the eight (8) legs listed in a) through h) selected by the instructor. The first leg will include ONLY normal procedures. The second leg may include emergency procedures.

L. Completion Standards: The pilot must perform all procedures and maneuvers to the tolerances listed in the Airline Transport Pilot/Type Rating Practical Test Standards. It is expected, during the earlier simulator flights, that the tolerance for completion of a maneuver or procedure is greater than during the later simulator flights. In all cases, it is expected that the pilot should strive to meet the tolerances listed below.

1. Takeoff
   - a. Heading ±5°
   - b. Airspeed ±5 KIAS
2. Departure, Cruise, Holding, and Arrival
   - a. Altitude @100 Feet
   - b. Heading ±10°
   - c. Airspeed ±10 KIAS
3. Steep Turns
   - a. Altitude ±100 Feet
   - b. Rollout Heading ±10°
   - c. Bank Angle ±5°
   - d. Airspeed +10 KIAS
4. Approach to Stall
   - a. Recognize Perceptible Stall or Stall Warning
   - b. Recover at First Indication of Stall
   - c. Strive for Minimum Altitude Loss
5. IFR Approaches (Prior to Final Approach)
   - a. Nonprecision
     - (1) Altitude ± 100 Feet
     - (2) Heading ±5°
     - (3) Airspeed +5 KIAS
   - b. Precision
     - (1) Altitude ±100 Feet
     - (2) Heading ±5°
     - (3) Airspeed ±10 KIAS

6. IFR Approaches (During Final Approach)
    a. Nonprecision Approach
        (1) CDI .... ±1/2 Scale Deflection
        (2) RMI .... ±5° Deviation
        (3) Bearing Pointer .... ±5° Deviation
        (4) MDA .... +50, –0 Feet
        (5) Airspeed .... +5, –0 KIAS
    b. Precision Approach
        (1) Glide Slope .... ±1/4 Scale Deflection
        (2) Localizer .... ±1/4 Scale Deflection
        (3) DH .... 0 Feet Prior to Initiating Missed Approach
        (4) Airspeed .... +5, –0 KIAS
7. Circling to Land
    a. Altitude .... +100, –0 Feet
    b. Heading .... ±5°
    c. Bank Angle .... Should Not Exceed 30°
    d. Airspeed .... ±5 KIAS
8. Missed Approach
    a. Altitude .... +100 Feet
    b. Heading .... ±5°
    c. Airspeed .... ±5 KIAS
9. Landing
    a. Final Airspeed .... +5, –O KIAS
10. Powerplant Failure—Multiengine Aircraft
    a. Altitude .... ± 100 Feet
    b. Heading .... ± 10°
    c. Airspeed .... ± 10 KIAS

CHAPTER 1
AIRCRAFT GENERAL

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CHAPTER 1
AIRCRAFT GENERAL

INTRODUCTION

This training manual provides a description of the major airframe and engine systems installed in the Hawker 800 XP.

This chapter covers the structural makeup of the airplane and gives a general description of the systems. No material is meant to supersede or supplement any of the manufacturer's system or operating manuals.

The material presented has been prepared from the basic design data, and all subsequent changes in airplane appearance or system operation will be covered during academic training and in subsequent revisions to this manual.

Annunciator panel in this manual displays all light indicators, and page ANN-3 should be folded out for reference while studying this manual.

GENERAL

The Hawker 800 XP is a twin turbofan engined, low wing monoplane aircraft, certificated for worldwide operation and intercontinental range by day or night.

The aircraft is operated by two crew, Captain and First Officer. A third crew member may be carried to carry out in-flight attendants duties. The standard layout is an eight-seat executive configuration.

Digital avionic systems are installed for communication, navigation, auto flight control and flight management systems. A five-tube electronic flight instrument system present the crew with displays of flight data received from other systems. Each turbofan engine is controlled by a digital electronic engine control unit, backed up by a hydro-mechanical reversionary unit. Hydraulically-operated thrust reversers are fitted.

A single hydraulic system pressurized by engine driven pumps (one per engine) supplies pressure for landing gear and doors, nosewheel steering, wheel brakes, stall identification system, wing flaps, airbrakes and thrust reverser functions. Backup hydraulic supplies are provided to operate the wheel brakes, stall identification, thrust reversers, lowering landing gear, and retraction or lowering of the flaps.

The main electrical services are supplied with 28 VDC from a starter/generator per engine. Emergency power is provided by batteries. Where AC power is required, it is supplied by either engine driven alternators (208 V three-phase) or static inverters.

A fluid anti-icing system is installed along the leading edge of the wings and horizontal stabilizer. Hot air is used to keep the main engine air intake free of ice. Electric heating is provided for windshields, pitot heads, static plates, stall vane, and rudder bias struts, plus the engine intake temperature and pressure sensors PT2 and TT2.

A gaseous breathable oxygen system is installed for use by crew and passengers during emergency conditions. The system also provides a therapeutic supply.

Fire detection and protection systems are installed for the main engines and for the APU. Overheat detection is provided for the rear equipment bay.

Engine bleed air is tapped from the engines for the pressurization, air conditioning and rudder bias systems.

A gas-turbine APU is installed to provide DC electrical power and air conditioning.

Leading particulars of the aircraft are presented in Figure 1.1.

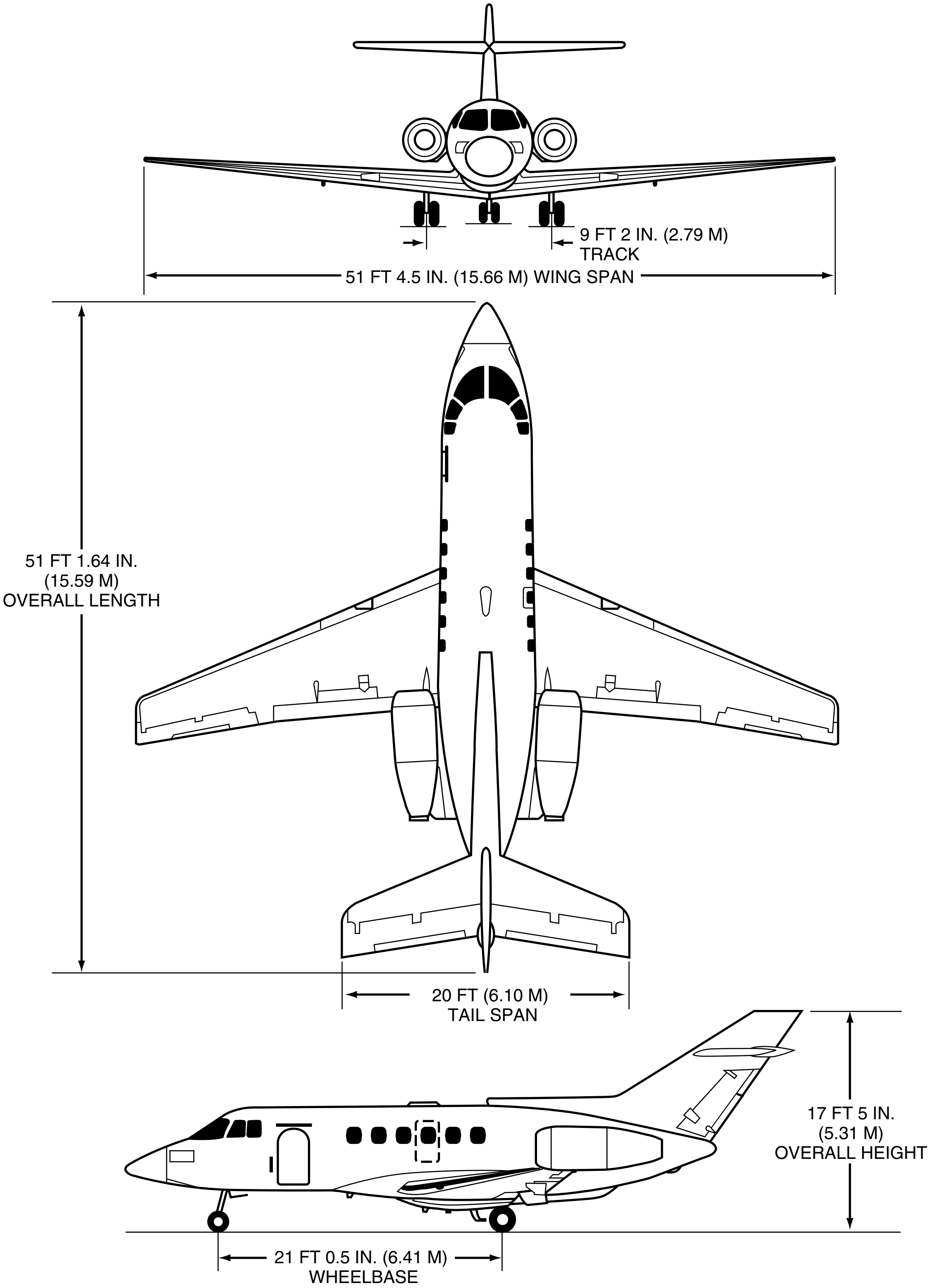


Figure 1-2. Airplane Dimensions

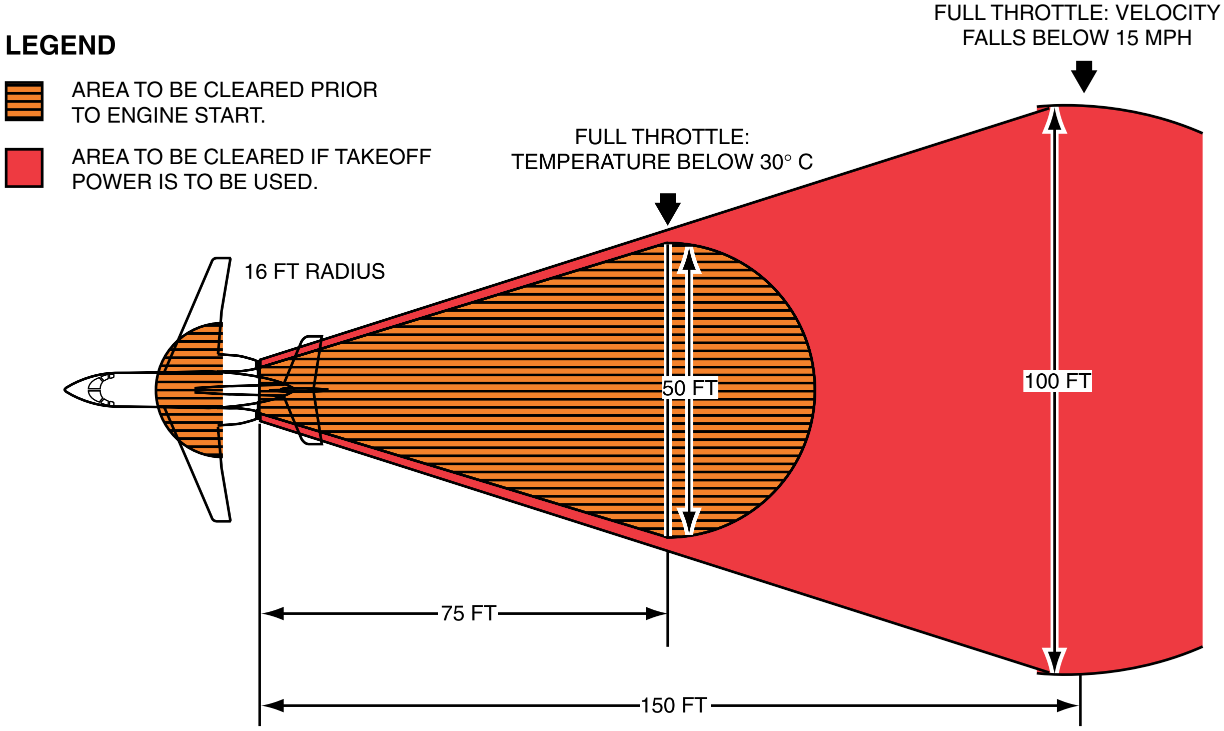
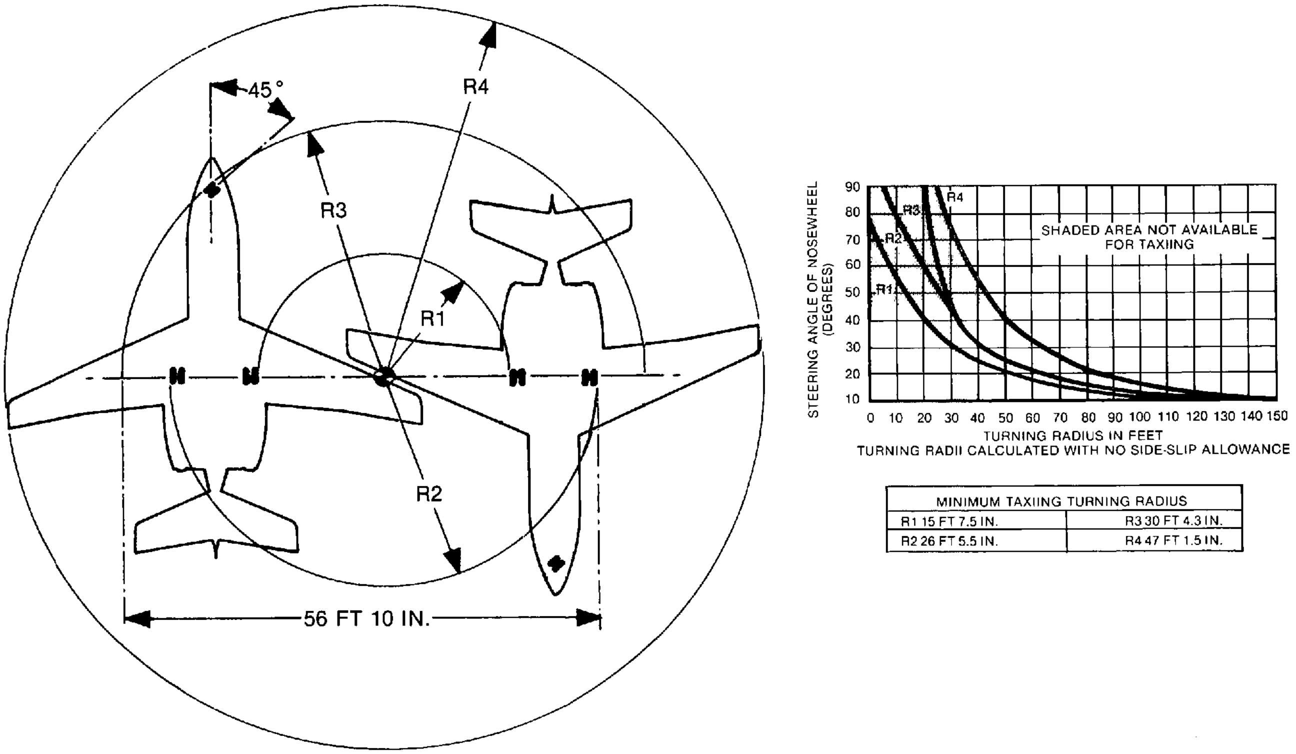

LEGEND
▤ AREA TO BE CLEARED PRIOR TO ENGINE START.
🟥 AREA TO BE CLEARED IF TAKEOFF POWER IS TO BE USED.
FULL THROTTLE: VELOCITY FALLS BELOW 15 MPH
FULL THROTTLE: TEMPERATURE BELOW 30° C
16 FT RADIUS
50 FT
100 FT
75 FT
150 FT

Figure 1-2. Danger Zones


45°
R4
R3
R1
R2
56 FT 10 IN.
STEERING ANGLE OF NOSEWHEEL (DEGREES)
90
80
70
60
50
40
30
20
10
R4
R3
R2
R1
SHADED AREA NOT AVAILABLE FOR TAXIING
0
10
20
30
40
50
60
70
80
90
100
110
120
130
140
150
TURNING RADIUS IN FEET
TURNING RADII CALCULATED WITH NO SIDE-SLIP ALLOWANCE

MINIMUM TAXIING TURNING RADIUS	
R1 15 FT 7.5 IN.	R3 30 FT 4.3 IN.
R2 26 FT 5.5 IN.	R4 47 FT 1.5 IN.

Figure 1-3. Minimum Turning Radius

FUSELAGE

General

The fuselage sections are the nose, center, and rear. These sections are further subdivided for discussion into the nose equipment bay, the cockpit, vestibule, passenger compartment, and the rear equipment bay. These are shown in Figure 1-4.

Nose Section

The nose section includes the area between the radome and the forward pressure bulkhead. It houses the radar and other equipment, and the nose wheel well (Figure 1-5). Access is through the hinged radome or the nosewheel doors.

Nose Compartment Ventilation System

Air is exhausted from the nose compartment during ground operation of the aircraft. If the exhaust fan fails or is blocked, an amber EFIS FAULT annunciator on the pilot's instrument panel and amber NOSE FAN on the pilot's transfer panel will illuminate. The EFIS FAULT annunciator can be canceled by pressing the face of the annunciator.

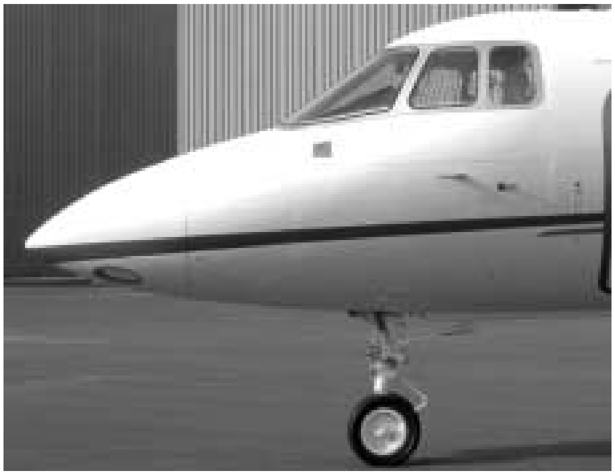


Figure 1-5. Nose Section

Pressurized Center Section

The pressurized center section extends from the forward pressure bulkhead to the aft pressure bulkhead and includes the cockpit, vestibule, and passenger cabin (see Figure 1-4). The cockpit seats two pilots and is fitted with a large, curved windshield in front of each position. The main entry door is located in the vestibule. The front baggage compartment is located on the right side of the vestibule. The No. 4 window on the right side of the cabin serves as an emergency exit.

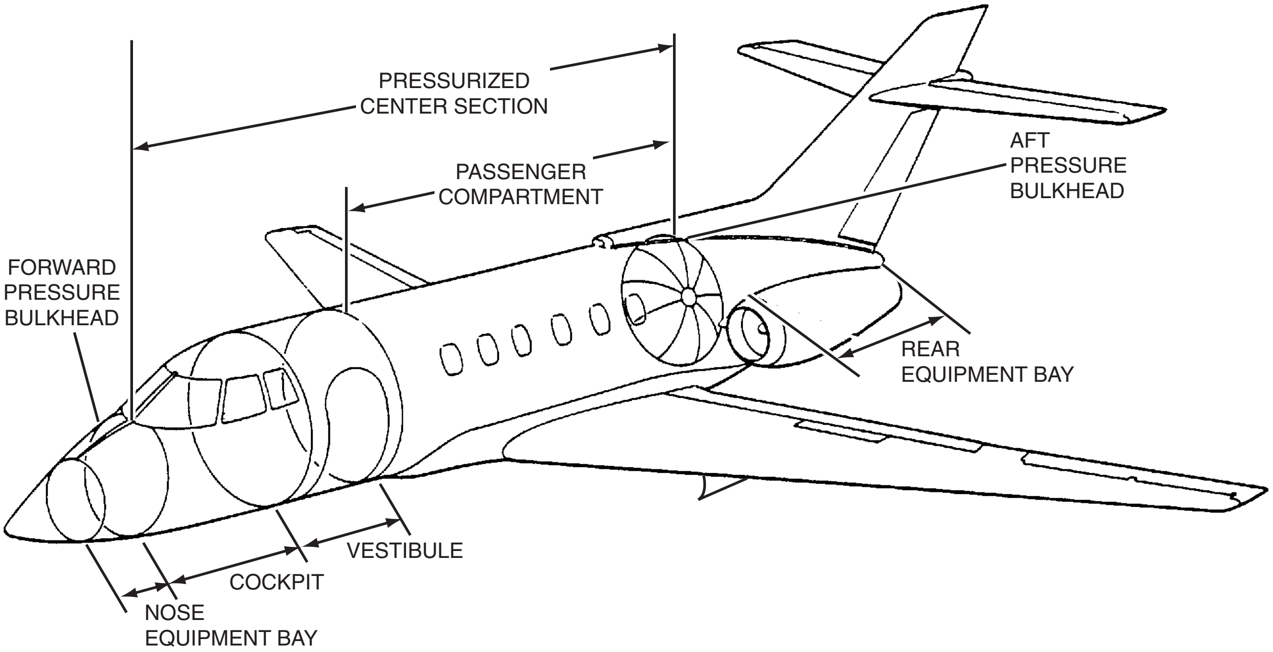


Figure 1-4. Fuselage Sections

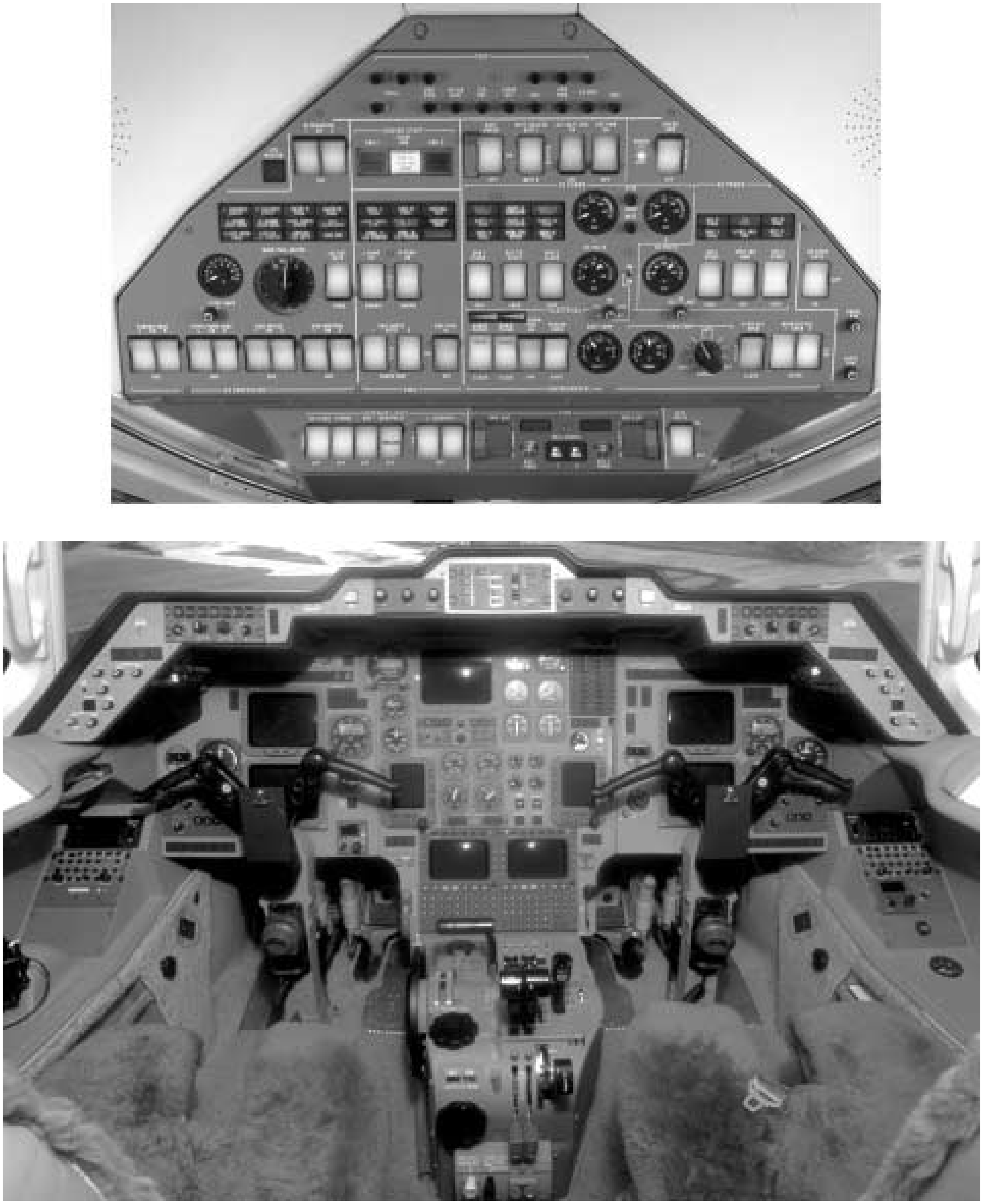


Figure 1-6. Cockpit Layout (Typical)

Cockpit

The general layout of the cockpit is shown in Figure 1-6, and a typical flight compartment is shown in Figure 1-7. Some instruments and equipment in the cockpit are installed to meet customer requirements and may vary from standard configurations.

The cockpit includes two IPECO adjustable crew seats. Each crew seat is mounted on rollers for fore and aft movement. A lever under the inboard side of the seat allows fore and aft seat positioning (Figure 1-8). Vertical adjustment is controlled by an outboard lever.

The thigh pad control knob raises or lowers the thigh pads. When the pilot uses the foot controls, the pressure on the thigh pad will overcome the spring loading, allowing the pad to move downward. When the pressure on the thigh pad is relaxed, the pad returns to its preset condition.

The recline control handle is pulled upward to allow the seat backrest to be pushed to the rear; releasing the control locks it in position. If the control is again pulled upward and pressure released from the backrest, it will return, under spring pressure, to an upright position and is locked upon release of the control handle.

The armrests can be adjusted for height by turning a control knob on the forward end. The inboard armrest can be stowed by rotating the armrest to the rear of the seat. When fully folded, the armrest is pushed in toward the center of the seat, thus reducing the seat width. The outboard armrest can be raised to an almost vertical position, where a friction clutch holds it in position until returned to the horizontal position by the occupant.

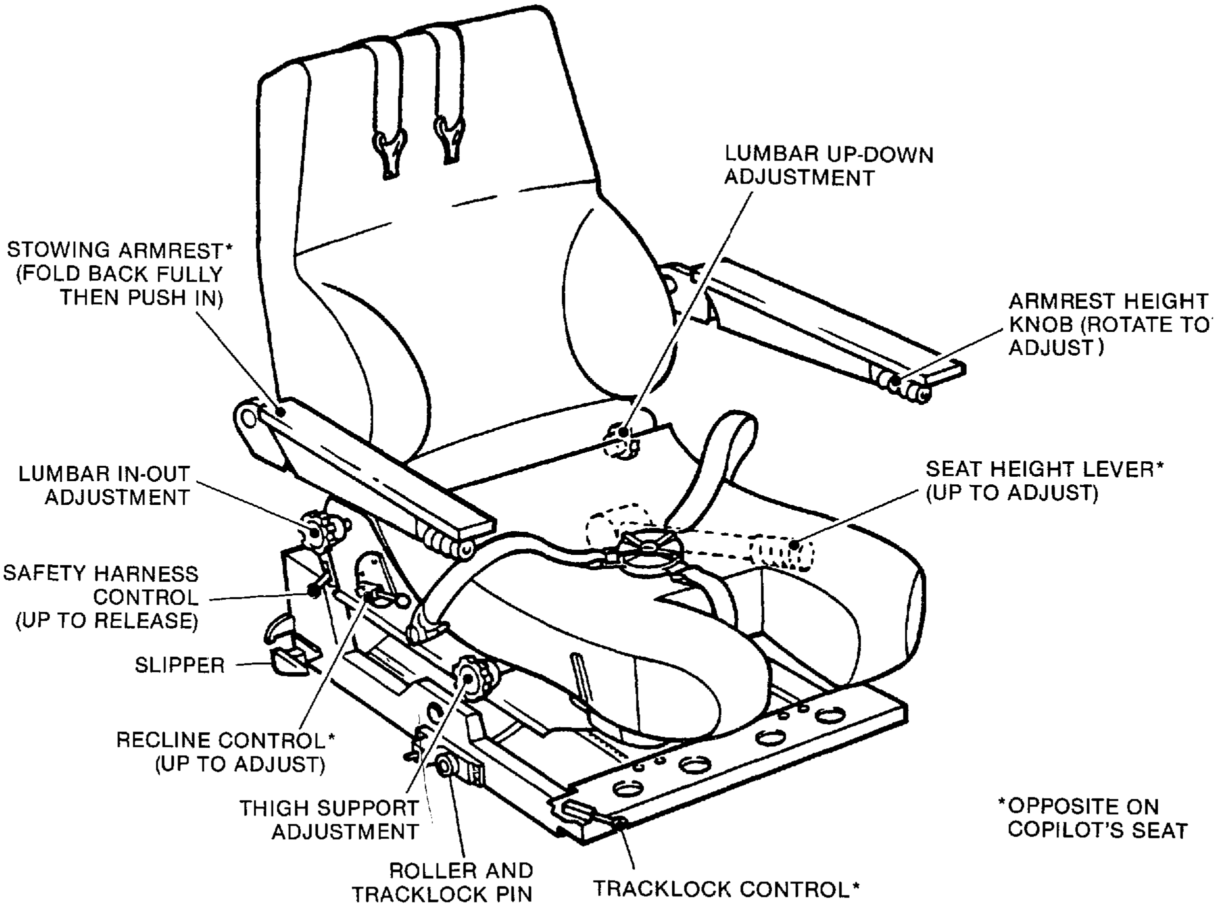

LUMBAR UP-DOWN ADJUSTMENT
STOWING ARMREST* (FOLD BACK FULLY THEN PUSH IN)
ARMREST HEIGHT KNOB (ROTATE TO ADJUST)
LUMBAR IN-OUT ADJUSTMENT
SEAT HEIGHT LEVER* (UP TO ADJUST)
SAFETY HARNESS CONTROL (UP TO RELEASE)
SLIPPER
RECLINE CONTROL* (UP TO ADJUST)
THIGH SUPPORT ADJUSTMENT
ROLLER AND TRACKLOCK PIN
TRACKLOCK CONTROL*
*OPPOSITE ON COPILOT'S SEAT

Figure 1-8. Pilot's Seat

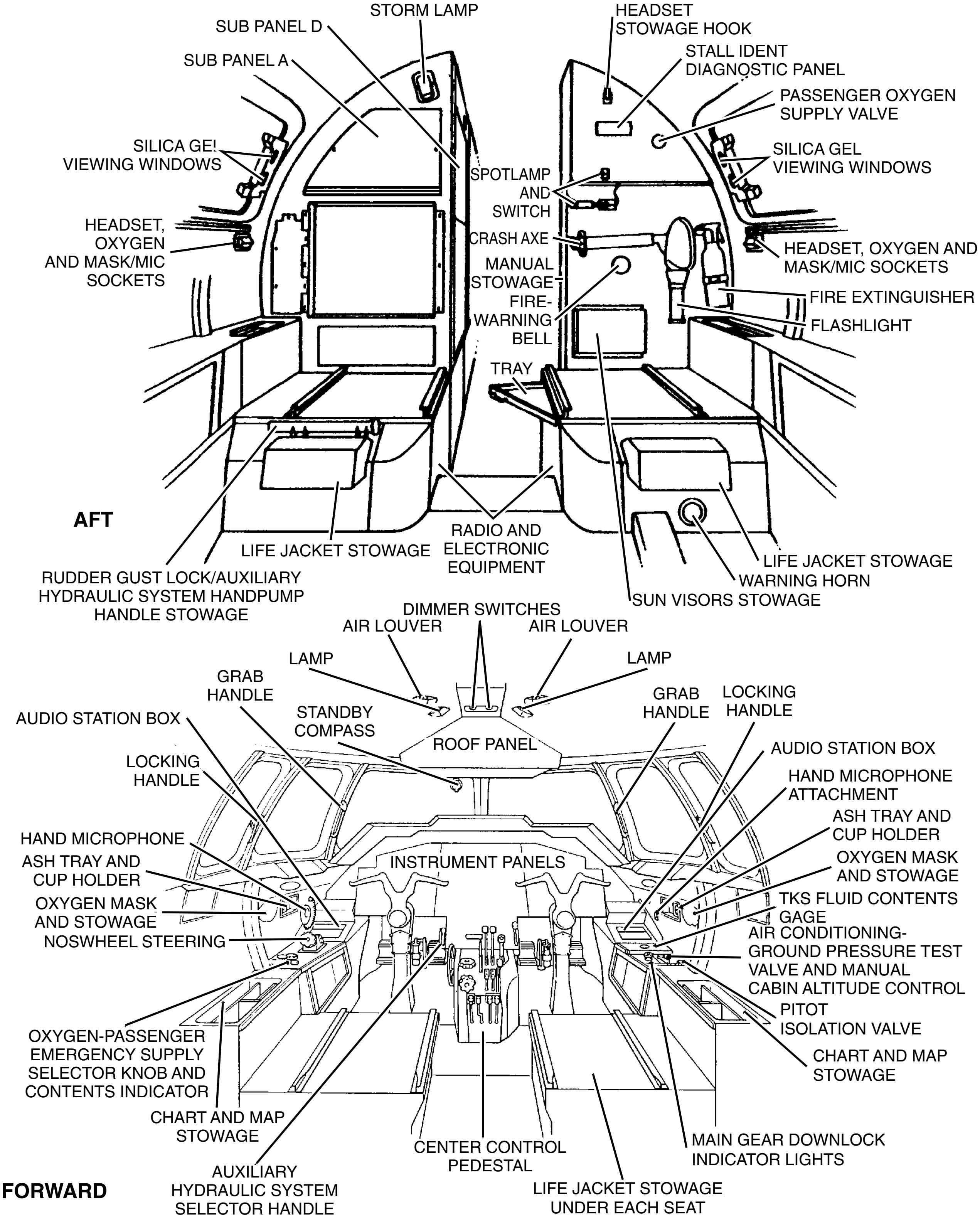


Figure 1-7. Flight Compartment (Typical)

The seat back cushion is adjustable for up-down and in-out lumbar support. In-out adjustment is controlled by a handwheel. Rotating it moves the lumbar cushion forward or backward. Up-down movement of the back cushion is controlled by a handwheel on the left side of the seat, rotation of which raises or lowers the back cushion to the desired position.

Vestibule

The vestibule contains the main entry door, a slide-away flight check seat, and the front baggage compartment (luggage bay).

The flight check seat is stowed forward of the avionics rack below the luggage bay. To use, the seat must be pulled out fully and the backrest raised to latch the seat in position. Seat occupancy is approved for takeoff and landing.

The baggage compartment has a capacity of 29.6 cubic feet. The bay is enclosed by a retaining net secured by zip fasteners and covered with a screen curtain. Equipment stowages and food modules are optional.

Standard factory-installed forward and rear cabinet units form the cockpit left bulkhead and the cabin left forward bulkhead. The main entry door separates the forward and rear cabinets.

The forward cabinet (Figure 1-9) contains an APU control panel and has space for additional equipment. An upper compartment is used for stowing therapeutic oxygen masks and a first aid kit. A lower compartment provides stowage for airplane manuals and documents. On the lower rear face of the cabinet, a small door provides access to the airframe deicing fluid tank filler. Below this access is a stowage compartment for landing gear ground locking pins, pitot head covers, static vent plugs, stall vent plugs, and other small items of ground equipment. A refreshment cabinet forms the rear cabinet unit.

A passenger cabin door separates the vestibule from the passenger cabin. This door must be secured in the open position for takeoff and landing. The door is held open by a magnetic latch.

Passenger Cabin

Cabin interior configurations vary widely with customer preference in the choice of equipment and number and arrangement of passenger seats. Figure 1-10 shows a typical eight-seat arrangement.

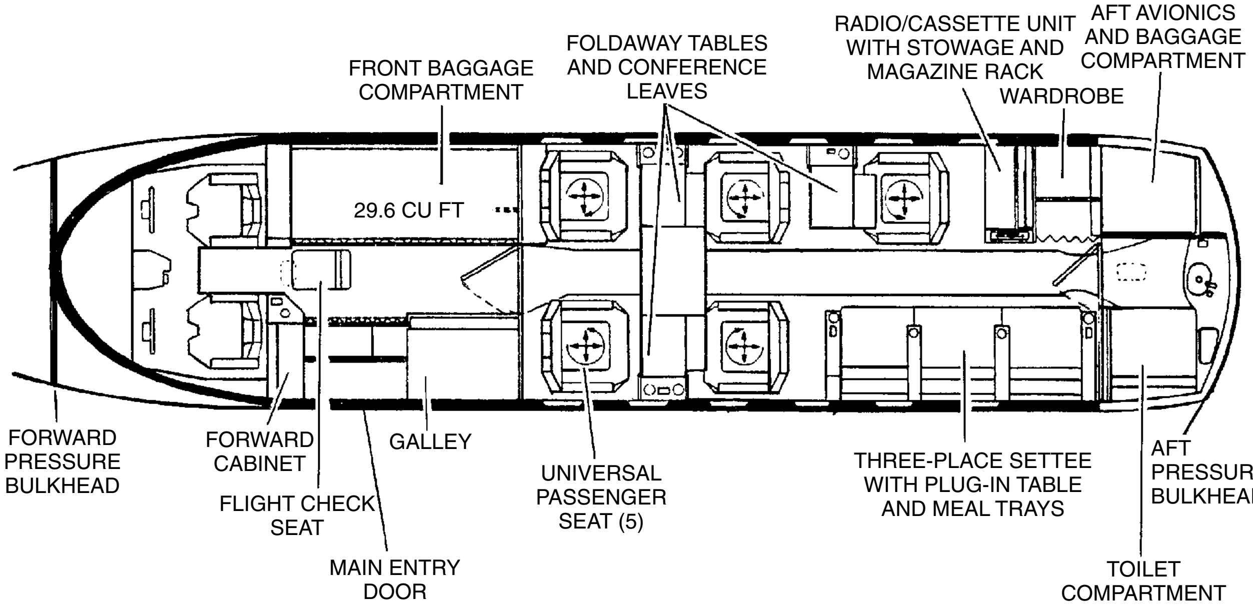

FRONT BAGGAGE COMPARTMENT
FOLDAWAY TABLES AND CONFERENCE LEAVES
RADIO/CASSETTE UNIT WITH STOWAGE AND MAGAZINE RACK
AFT AVIONICS AND BAGGAGE COMPARTMENT
WARDROBE
29.6 CU FT
FORWARD PRESSURE BULKHEAD
FORWARD CABINET
FLIGHT CHECK SEAT
MAIN ENTRY DOOR
GALLEY
UNIVERSAL PASSENGER SEAT (5)
THREE-PLACE SETTEE WITH PLUG-IN TABLE AND MEAL TRAYS
AFT PRESSURE BULKHEAD
TOILET COMPARTMENT

Figure 1-10. Flight Compartment (Typical)

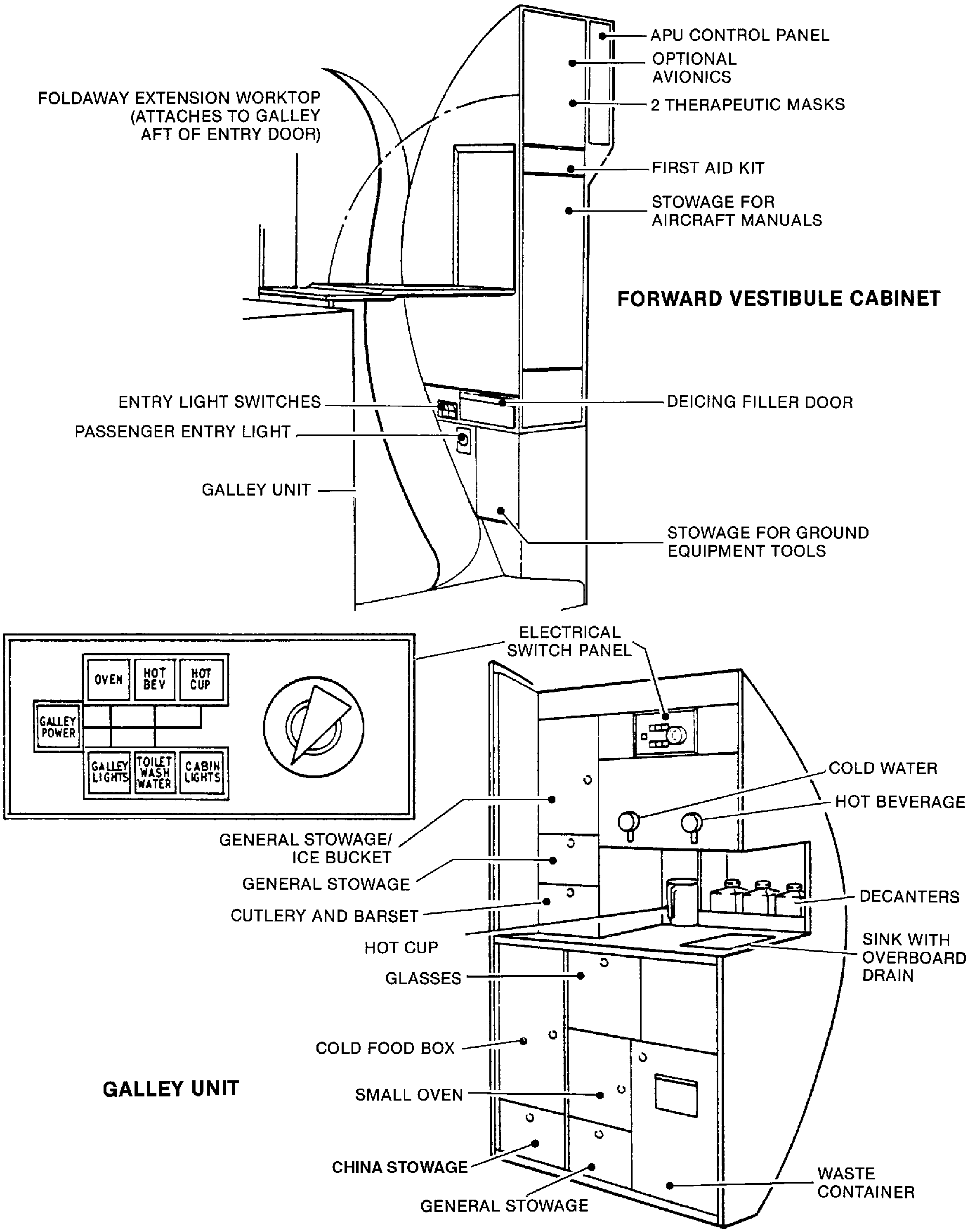


Figure 1-9. Vestibule Arrangement

Each forward- or aft-facing seat has an adjustable backrest controlled by a knob on the outboard armrest and a retractable inboard armrest with a control knob. With the knobs held in the up position, the mechanism is unlocked, enabling the rests to be repositioned as required.

Passenger service units, installed on each side of the cabin roof, provide individual reading lights and controllable air outlets. Roller blinds are installed at each of the 12 cabin windows. Carpets are provided to completely cover the floor area and lower side walls.

At the right-hand rear of the passenger cabin is a radio/cassette unit with stowage below. Immediately above this unit is a magazine rack which is attached to the forward face of a wardrobe equipped with a coat rack, a switch and interior light, stowage for conference tables leaves, and closed by a sliding curtain.

Main Entry Door

The main entry door is located on the left side of the vestibule (Figure 1-11). It is one-piece and opens outward and down. Integral stairs and handrail are provided.

The door is counterbalanced by a spring-loaded tensator motor and secured by external and internal handles. The external handle, installed flush with the fuselage and forward of the door, has a key lock. The internal handle is mounted forward of the door opening. Provision is made on the door for attaching a custom's padlock. A rubber seal is installed around the door to prevent cabin pressure leakage. It is self-energizing, forming an airtight seal when the cabin is pressurized. The higher the pressure differential, the greater the sealing effect.

The door is opened either by moving the internal handle upward from the LOCKED position to the UNLOCKED position or by moving the external DOOR PULL handle fully downward. The initial movement of either handle unlocks the door which then lowers under the braking effect of the tensator (spring/friction) motor.

The door is closed by pulling the handrail inward. This action breaks a geometric lock in the door stay joint and raises the door, assisted by the tensator motor. The door is then locked in position by either handle.

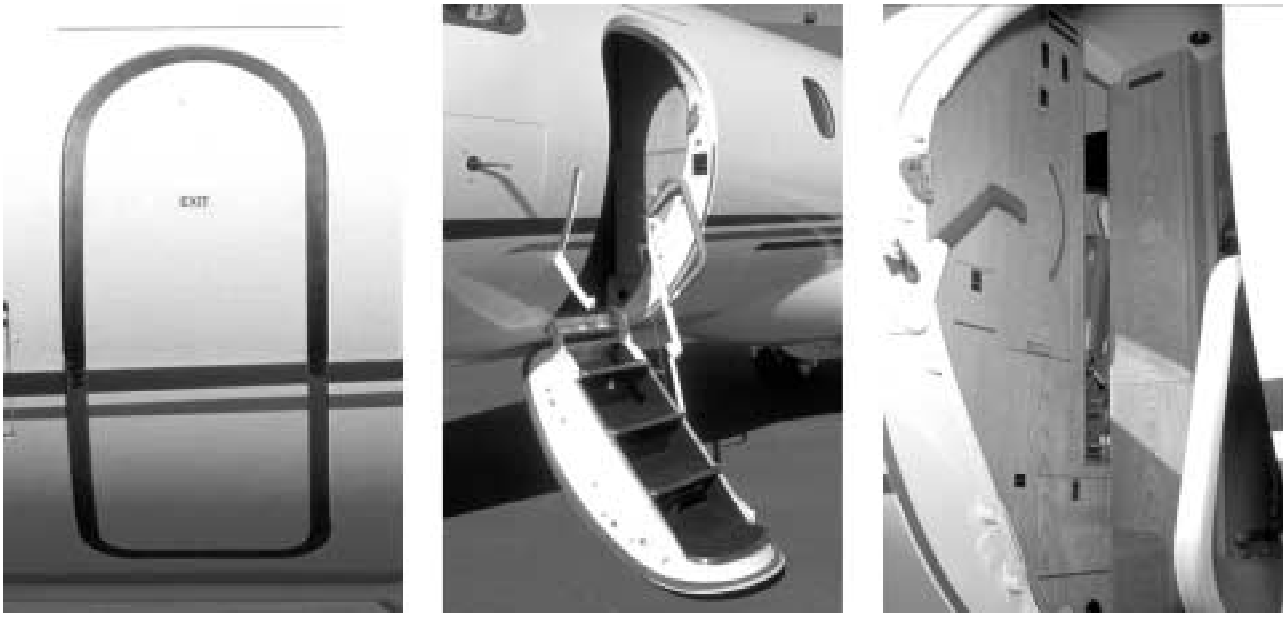


Figure 1-11. Main Entry Door

Two microswitches, one installed on the doorframe and the other in the door operating mechanism, are connected to the master warning system ENT DOOR UNLOCKED annunciator. Either switch lights the annunciator when the door is not in a fully locked position. The DC power supply to the microswitches is taken from bus bar PE.

Toilet Door

An outward opening solid door is provided for the toilet compartment. The door is locked shut from the inside by a shoot bolt.

Escape Hatch

On the right side of the fuselage is an inward opening emergency exit escape hatch (Figure 1-12). It incorporates the fourth window on that side, over the wing root. It also incorporates a rubber seal for pressurization retention. The exit may be opened either by a PULL handle internally or by a PRESS button on the outside.

The internal release handle may be modified to accept a ground security latch. This exit is an emergency rescue access point, so the ground security latch incorporates a REMOVE FOR FLIGHT flag.

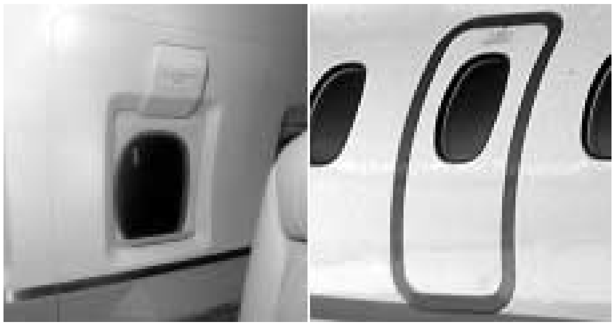


Figure 1-12. Escape Hatch

Emergency Equipment

Following is a list of standard and optional emergency equipment available on the Hawker 800 XP (Figure 1-13).

- Hand fire extinguisher—One on the left bulkhead of the cockpit. Optional hand fire extinguisher in the cabin.
- Smoke goggles—Stowed on cockpit side walls or side consoles
- Therapeutic oxygen masks—Two stowed in the forward vestibule cabinet (for use with the airplane oxygen system)
- Crash axe—One on the left bulkhead of the cockpit
- Flashlights—One on the left bulkhead of the cockpit
- First aid kit—Stowed in the forward vestibule cabinet. The stowage is labeled “FIRST + AID.”
- Lifejackets (standard airplane)—
  - One at the front of each pilot’s seat base
  - One in the forward vestibule cabinet
  - One under each armchair
  - Four in the forward armrest of the three-seat settee
- Smoke mask—One in the cockpit right-side console (for use with portable oxygen)
- Portable oxygen—One bottle stowed on the right bulkhead of the cockpit

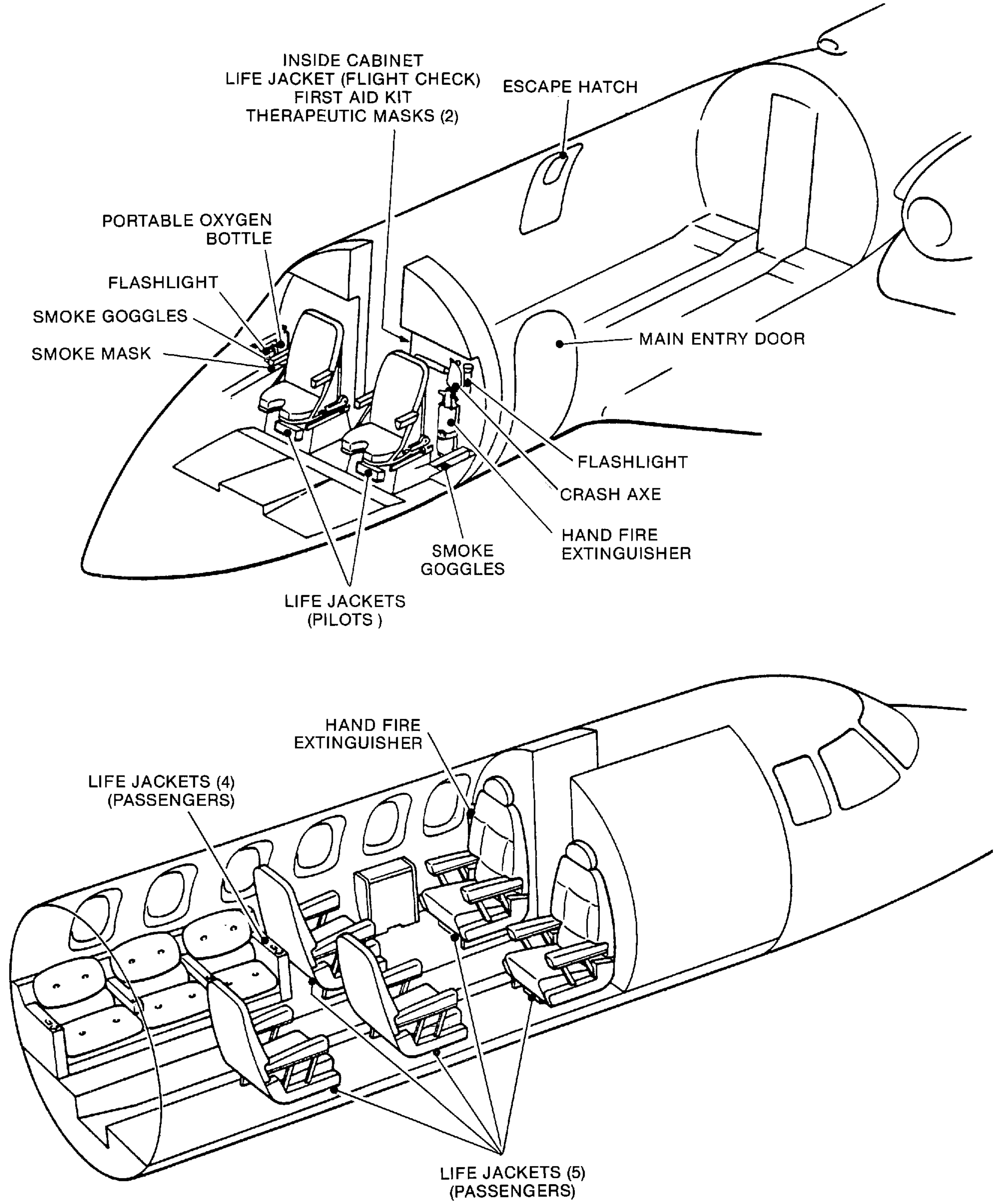


Figure 1-13. Emergency Equipment Locations

Windows

There are six windows in the cockpit, three each on the pilot's and copilot's sides. The windshield and forward side windows are fixed; the aft side windows are hinged at the rear.

The windshields and forward side windows in the cockpit (Figure 1-14) are made from two clear, stretched acrylic plastic plies and an outer ply with a heating element applied to its inner face. (See Chapter 10, "Ice and Rain Protection," for more information.)

The aft cockpit side windows and the passenger cabin windows are constructed of three plies of acrylic plastic. Air gaps between the plies of the aft cockpit windows are connected to a silica gel container at the rear of each window (Figure 1-15). All cabin windows (see note) share a common silica gel container mounted in the forward baggage compartment.

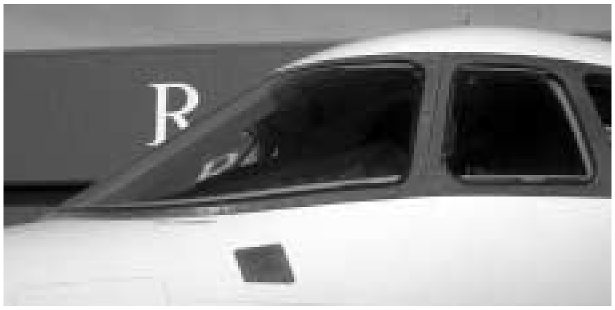


Figure 1-14. Heated Windows

NOTE

The window installed in the escape hatch is not connected to a common silica gel container. It has its own container.

The aft cockpit side windows open inward. A locking latch is provided at the forward edge of the window.

Extreme care must be taken when cleaning and polishing the acrylic plastic windows. To avoid crazing (caused by frictional heat), all polishing must be done by hand. Using a soft cloth to carry the cleaning solution to the window, rinse thoroughly.

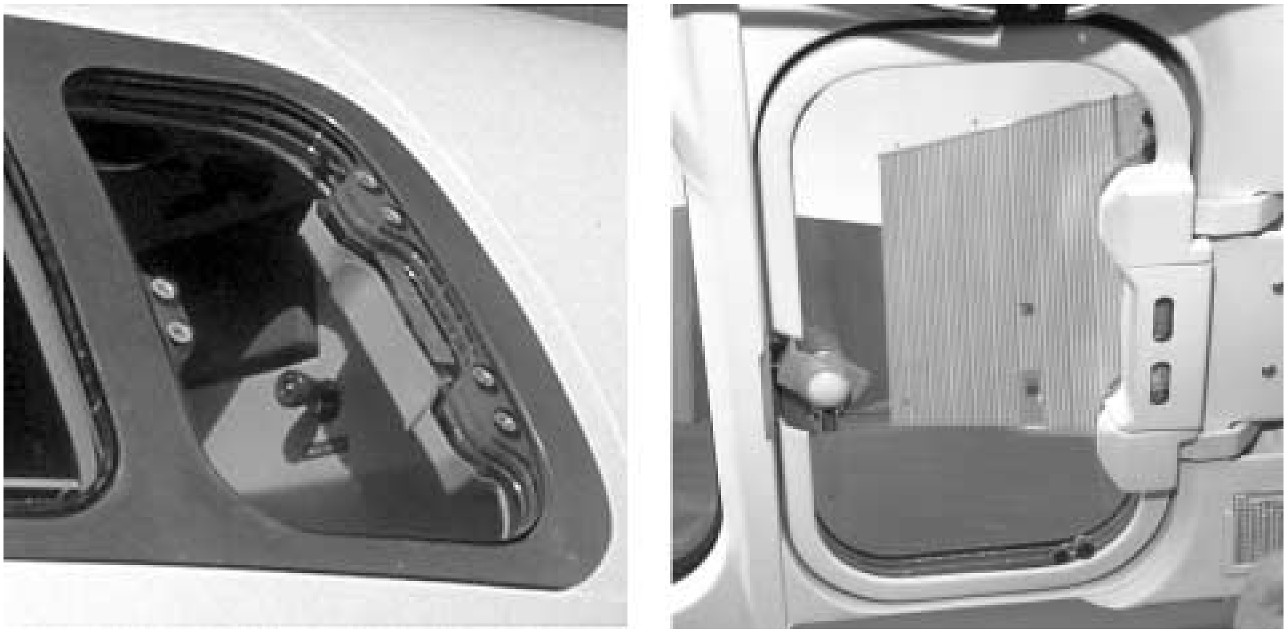


Figure 1-15. Cockpit Aft Side Window

Rear Section

The rear section includes the rear equipment bay, a tail cone, and attachment points for the vertical stabilizer, ventral fuel tank, and the engine pylons. A hinged door located in the bottom of the ventral tank provides access to the unpressurized rear equipment bay (Figure 1-16).

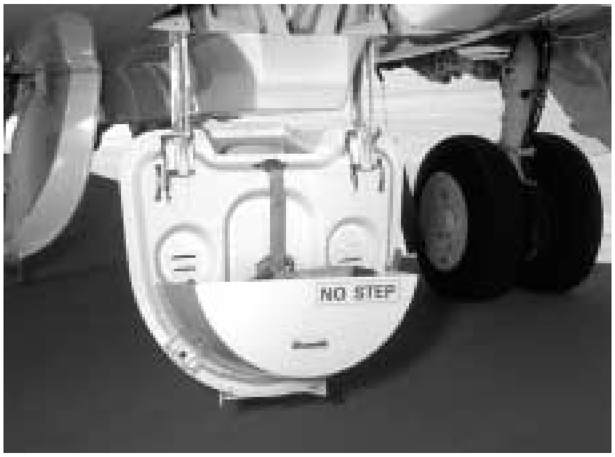
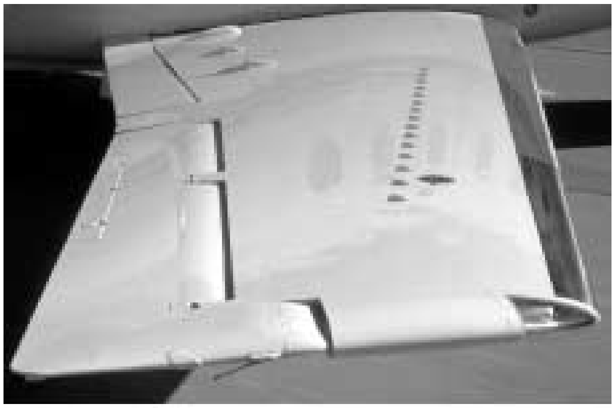


Figure 1-16. Rear Equipment Bay Hatch

WING

The Hawker 800 XP has a swept-back, cantilevered, all-metal wing (Figure 1-17). The center wing section is contoured to fit the fuselage and is attached to the fuselage at four attachment points.

Air flow control devices on each wing consist of a row of vortex generators, a vortilon and a stall trigger. The vortex generators are attached to the upper surface of the wing. The vortilon is on the bottom surface of the wing leading edge, inboard of the inner end of the row of vortex generators. The stall trigger is attached to the anti-icing distributor inboard of the landing lamp.



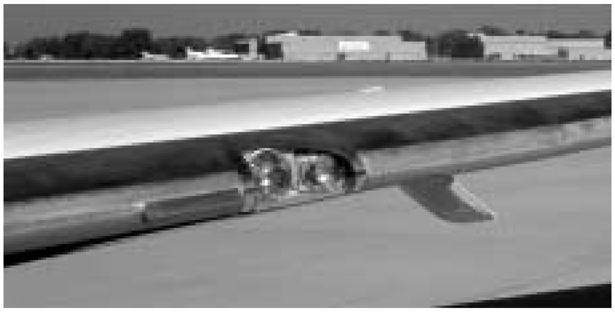


Figure 1-17. Wing

On the trailing edge a double slotted flap extends from the wing to the fuselage. An upper and lower air brake is positioned forward of the flap; in the closed position the air brakes form part of the flap shroud.

The major portion of each wing is sealed to form an integral fuel tank. Top skins are machined single sheets from root to tip. The bottom skins are machined single sheets from airplane centerline to tip. The leading edge of each wing contains the landing and taxi lights and deicing fluid panels.

EMPENNAGE

The empennage (Figure 1-18) consists of the horizontal stabilizer, elevators with trim tabs, vertical stabilizer, and a rudder with trim tabs. The horizontal stabilizer is mounted high on the vertical fin away from airflow disturbance caused by the engine exhausts. Both the vertical and horizontal stabilizers are all-metal, using spars and stressed-skin construction. A ram-air scoop is housed in the dorsal fin forward of the vertical stabilizer.

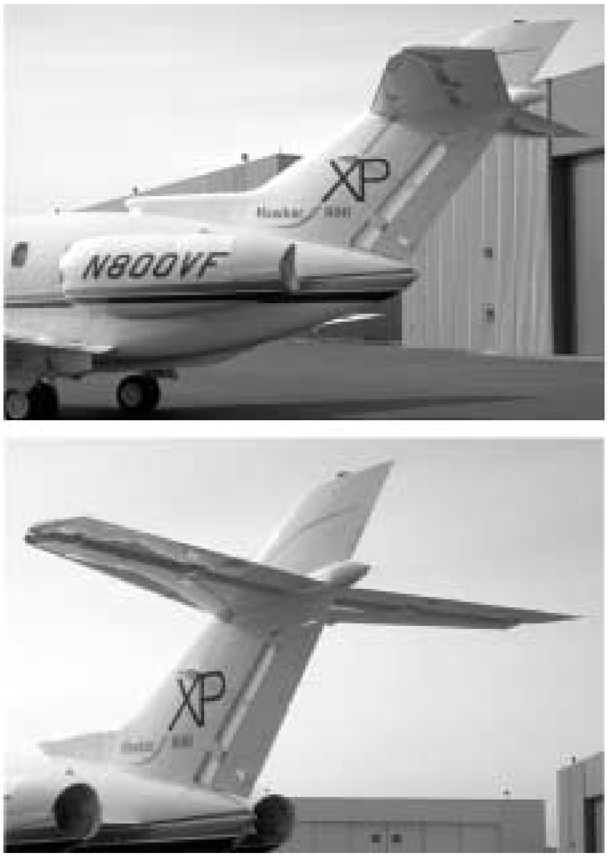


Figure 1-18. Empennage

AIRPLANE SYSTEMS

GENERAL

The following is a brief introduction to the major airplane systems on the Hawker 800 XP. Detailed descriptions of these systems are contained within the individual chapters of the training manual.

ELECTRICAL POWER SYSTEMS

The airplane electrical system is a 28-VDC system which receives power from two starter-generators connected through a split bus bar system. An additional starter-generator is available from the APU. Two main batteries provide normal electrical system backup power and a power source for internal engine starting. An external power unit can be connected via a three-pin plug to energize the bus bars for servicing and engine starting. Two main inverters provide single-phase, 115-volt, 400-Hz AC power. A standby inverter will supply essential AC-operated systems in the event of failure of both main inverters. Two transformers provide 6-and 26-volt AC power. Two engine-driven alternators provide independent 208-VAC power for windshield, sidescreen, and stall vane heating. Two DC ammeters, a DC voltmeter, and an AC voltmeter are used to monitor the electrical system.

LIGHTING

The Hawker 800 XP has standard navigation, beacon, strobe, landing, and taxi lights. A left and right wing ice light is installed for wing ice inspection. Interior lighting includes instrument illumination and passenger cabin lamps, step and vestibule lamps, passenger reading lamps, toilet compartment lamps, and galley and cabin notice lamps. Optional logo lamps are available.

MASTER WARNING SYSTEM

The master warning system (MWS) panel is located on the center instrument panel. Ice protection, fuel, and electrical system annunciators are grouped on the roof panel. Roof and MWS panel annunciators are red, amber, and white. Red warnings are those which require immediate crew action. Amber and white annunciators denote faults or system conditions of less urgency to the crew. Test buttons are provided to test all annunciator filaments and warning circuits.

FUEL SYSTEM

The Hawker 800 XP fuel system includes one integral (wet) fuel tank in each wing and a metal ventral tank in the lower aft fuselage. The system has both a wing tank-to-tank transfer capability and a crossfeed capability from either wing tank to one or both engines. One electric boost pump in each wing tank delivers fuel under pressure to its respective engine during normal operating conditions. Each boost pump operates two jet pumps for transfer of fuel from outer portions of the same tanks and from the opposite wing tanks and ventral tank. The fuel system has both gravity- and pressure-refueling/defueling capabilities. All tanks can be pressure-refueled or defueled simultaneously.

AUXILIARY POWER UNIT

The APU is a self-sufficient unit installed in the rear equipment bay that renders the airplane independent of ground service equipment. Both the Garrett Turbine Engine Company (AiResearch) and Solar Turbines Incorporated, (Turbomach) manufacture power units installed in the Hawker 800 XP. Its function is to drive a DC generator to power the aircraft electrical system, charge the aircraft batteries, and provide bleed air to the air conditioning system for heating and cooling.

POWERPLANT

The aircraft is propelled by two Garrett AiResearch Model TFE 731-5BR turbofan engines installed in pods mounted on pylons; one on each side of, and integral with, the rear fuselage.

Firewalls divide each pod into two fire zones which are ventilated by ram air; both zones incorporate a fire/overheat warning system. The two-shot fire extinguishing system discharges only into zone 1, the forward zone.

Hot air is bled from the engine to pressurize and air condition the aircraft, to operate the rudder bias system and for engine anti-icing.

Each engine has a combined starter/generator and can be started from either the aircraft batteries or a ground power supply. Each engine has an AC generator for the pilot's windshield de-icing.

Engine power and fuel shutoff controls for each engine are operated by separate throttle and high pressure cock levers on the pilot's central control pedestal.

Engine starting, ignition and anti-ice controls are located on the flight compartment roof panel. Instruments (i.e. N_1, N_2 rpm, ITT, oil pressure and temperature indicators) are located in the center instrument panel. Annunciators associated with the engine are on the main MWS and roof panel.

The Hawker 800 XP airplanes are equipped with a Dee Howard TR5000BA Thrust Reverser System. The system includes a thrust reverser assembly at the rear of each engine nacelle and various controls and indicators located within the cockpit.

FIRE PROTECTION

The Hawker 800 XP fire protection system includes the detection and warning of fire and overheat conditions in each engine and within the APU. Overheat detection capability is also provided for the rear equipment bay. Fire extinguishers are installed to extinguish engine and APU fires. The engine fire extinguishing system is a two-shot system; if an engine fire is not extinguished with actuation of the first bottle, the second bottle is available for discharge into the same engine. The engine fire bottles are located in the rear equipment bay. The APU has independent fire detection, warning, and extinguishing systems. A portable handheld extinguisher is located in the cockpit.

PNEUMATICS

The Hawker 800 XP uses engine bleed air to supply pneumatically operated airplane systems. Each engine has a low- and high-pressure bleed-air port. Low-pressure air is provided to the air-conditioning, pressurization, and rudder bias systems. Low-pressure air is also used for the air jet pump and to pressurize the hydraulic reservoir. Normal operation of the system introduces high-pressure air to supplement the low-pressure air supply to the air-conditioning system at low power settings to maintain minimum pressure requirements. Pneumatic controls are located on the roof panel, and fault annunciators are located on the center instrument panel.

ICE AND RAIN PROTECTION

The Hawker 800 XP is approved for flight into known icing conditions when the required equipment is installed and operational. Ice detection is provided by a rotary-cutter system and two wing ice-inspection lights. The wing and horizontal stabilizer leading edges are anti-iced by a TKS fluid system. The windshields and forward sidescreens are electrically heated to prevent icing. Windshield rain removal is accomplished by the aerodynamic shape of the windshield. Electric heating also is provided for the pitot heads, forward static plates, rudder bias struts, and airflow angle sensors. Engine cowlings are anti-iced by engine high-pressure bleed air. The PT2/TT2 sensor is electrically heated.

AIR CONDITIONING

Air supply for air conditioning is taken from the low- and high-pressure bleeds of each engine. The air passes through a ram-air heat exchanger and then a refrigeration (cooling turbine) unit. Precooling of engine bleed air is accomplished in the ram-air heat exchanger. The ram cooling air is provided through the dorsal intake duct. On the ground, airflow is induced through the heat exchanger by a fan driven by the cooling turbine. The bleed air then flows to the three-wheel cold air unit where the conditioned air temperature is regulated by the amount of air allowed to pass through the turbine. The more air that passes through the turbine, the cooler the temperature. The more air that is allowed to bypass the cooling turbine, the warmer the temperature. The conditioned air is then routed through a water separator to remove moisture. Either engine bleed air or APU bleed air can be used for the air-conditioning system when on the ground. Engine bleed only can be used when airborne. Full range manual control of temperature is available in the event of malfunction of the automatic features.

PRESSURIZATION

The cabin is pressurized by engine bleed air through the air-conditioning system. Cabin pressure is automatically controlled by two outflow/safety valves which are pneumatically operated by the difference between cabin and ambient pressures. These valves govern the exhausting of cabin air to the atmosphere. Safety devices completely shut off the outflow when the cabin altitude reaches a preset value. Full range manual control of pressurization is available in the event of malfunction of the automatic features. A dump valve is provided to control cabin pressure when normal system control is inoperative. When the dump valve is selected fully open, a ram-air valve opens to ventilate the unpressurized cabin.

HYDRAULIC POWER SYSTEMS

The Hawker 800 XP has both main and auxiliary hydraulic systems. The main system is pressurized by two engine-driven pumps, one on each engine. The system provides pressure for actuation of the landing gear, flaps, airbrakes, nosewheel steering, brakes, and stick pusher. System operation is monitored by annunciators and a system pressure gage. The auxiliary hydraulic system is pressurized by a hand pump. It provides pressure for emergency extension of the landing gear and for flap operation.

LANDING GEAR AND BRAKES

The Hawker 800 XP landing gear is hydraulically actuated and fully enclosed by hydraulically and mechanically actuated doors. Gear position and warning are provided by indicator lights and a warning horn. Nosewheel steering is also hydraulically powered. The brake system is controlled by master cylinder pressure and provides for normal and emergency braking. Antiskid braking is available during normal braking. Parking brakes are applied with a lever in the cockpit.

FLIGHT CONTROLS

Primary flight controls include ailerons, rudder, and elevators. All are mechanically actuated. Secondary flight controls include flaps and airbrakes which are powered by the main hydraulic system. The flaps also can be powered by the auxiliary hydraulic system. Trimming of the primary flight controls is mechanical in all three axes, with electric elevator trim available. A spring-loaded strut, compressed during rudder deflection, assists in return of the rudder to neutral. The rudder also incorporates a pneumatically actuated bias system to counter asymmetric thrust during single-engine operation.

PITOT-STATIC SYSTEM

The pitot-static system serves the computers of air data systems, a standby altimeter, a standby airspeed indicator, the cabin differential pressure gage, and stall warning and identification system. Provision is made for addition of optional equipment. Pitot pressure supplied to the stall detectors and additional equipment can be isolated by manual selection. Two pitot heads provide impact pressure, four static vent plates provide static pressure, and two wing vents provide stall vent pressure. Water traps and drains are provided in the pipelines. The pitot heads and forward static vent plates are heated to prevent icing.

AVIONICS

Standard avionics includes the EFIS 86 E or SPZ 8000 Electronic Flight Instrument Systems. The system consists of four identical electronic flight display units and one multifunction display unit together with the necessary processor units and control panels. Two of the four display units are installed one above the other on each main instrument panel. They replace the conventional electromechanical ADI and HSI on each side. The multifunction display unit occupies the weather radar position with a weather radar control panel mounted directly below the unit. In addition to information previously displayed on a conventional ADI and HSI, items such as air data, annunciation of flight director modes, waypoint and VOR/LOC locations, weather patterns, together with checklists and diagnostic messages, are also available. The flight director function can be used independently of the autopilot system with the pilot steering the airplane to satisfy the flight director commands as programmed, or the autopilot may be coupled to automatically steer the airplane to satisfy flight director commands as programmed. The yaw damper system operates independently of the autopilot and may be engaged with or without the autopilot engaged.

OXYGEN SYSTEM

The Hawker 800 XP oxygen system provides oxygen for crew and passenger use from two high-pressure steel cylinders. In flight, oxygen is available to the crew at all times. The passenger's oxygen is available automatically at high cabin altitudes or manually anytime at the pilot's discretion. Oxygen for therapeutic purposes is available at all times at selected cabin outlets. A portable oxygen system (smoke set) is stowed in the cockpit.

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ELECTRICAL POWER SYSTEMS

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CHAPTER 2
ELECTRICAL POWER SYSTEMS

INTRODUCTION

This chapter contains a description of the electrical power system used on the Hawker 800 XP, including information on the DC and AC systems. The DC system consists of storage, generation, distribution, and monitoring DC power. The AC system consists of generation, distribution, and monitoring of the constant-frequency (inverter) and the variable-frequency (alternator) AC systems. Also included in this chapter are specific limitations.

GENERAL

Basic electrical power of the Hawker 800 XP is provided by the DC system. This system consists of two 30-volt, 400-ampere brush-type starter-generators, one on each engine. An additional starter-generator is available from the auxiliary power unit (APU). Two main airplane batteries are provided. Two additional batteries are also provided, one for standby instrument power and the other for emergency avionics power. The DC supply from the generators and batteries has a positive polarity and uses the low-resistance bonded airframe as a return current path. Power from the two main generators and the APU generator is distributed to the airplane systems via a split bus bar system. A generator control unit (GCU) in each generator circuit stabilizes the bus bars at 28.0 volts. With the generators on line, manual selection of a BUS TIE switch enables the system to be paralleled or split into two single-channel systems. Automatic splitting occurs when a bus bar protection unit voltage sensor detects undervoltage or overvoltage conditions.

Nos. 1 and 2 batteries, normally connected in parallel, can be manually selected to provide DC supplies for essential services.

DC power distribution is through a split bus bar system (PS1 and PS2) and two essential system bus bars (PE and PE2).

With only internal power (airplane batteries) available, the split bus bars (PS1 and PS2) are not available. This arrangement is required in order to restrict loads on the batteries until such time as a generator (main or APU) comes on line. With any generator on line or with 28-volt external (ground) power connected, all DC bus bars are available.

NOTE

With only one generator on line, the bus tie must be closed to power both PS1 and PS2.

Inverters, powered from the airplane DC system, provide 115 volts AC for the operation of various airplane systems and navigation equipment.

Each of the two engine-driven alternator produces an independent 208-volt supply for windshield heating. The 7.4 KVA alternators are "frequency wild" (variable frequency), three-phase devices.

Figure 2-1 shows the electrical system components locations.

Figure 2-2 shows the circuit-breaker panel locations, and Figure 2-3 shows the main distribution panel

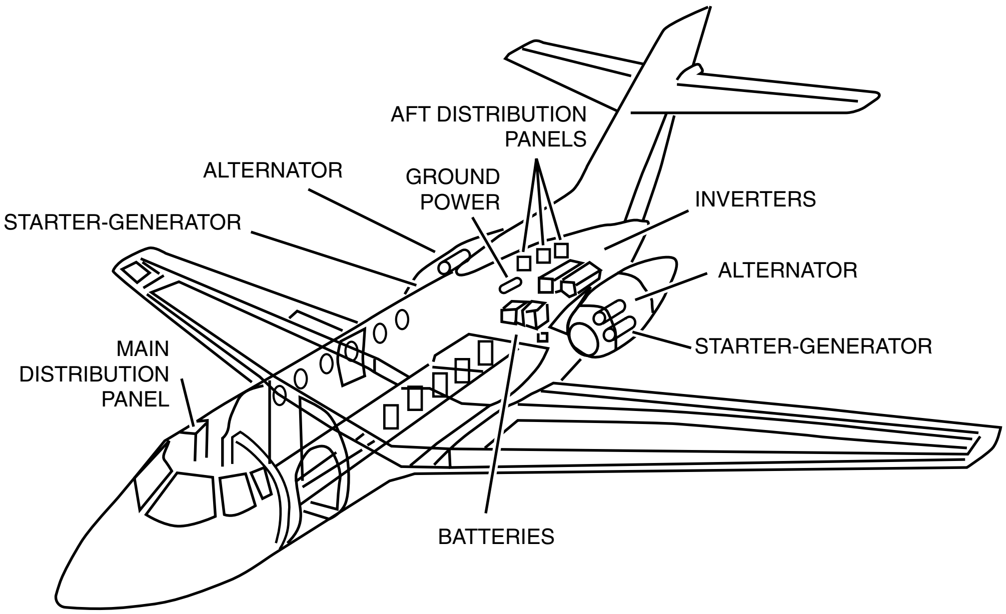


Figure 2-1. Component Locations

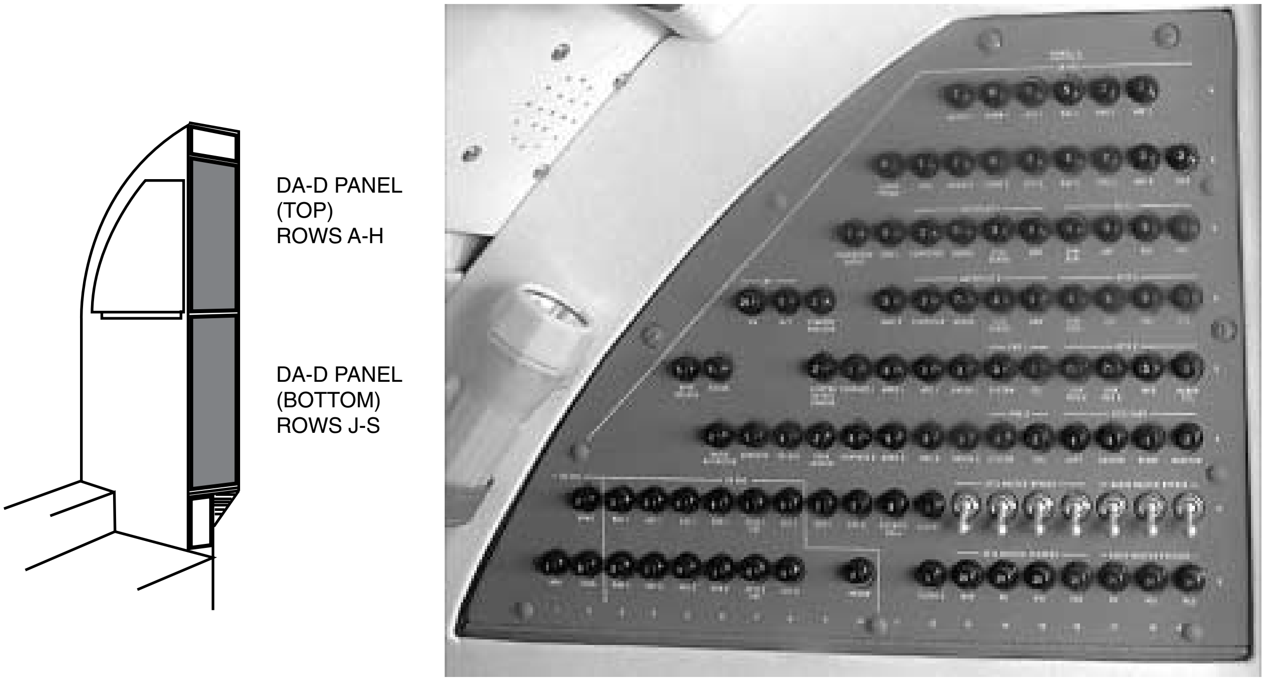
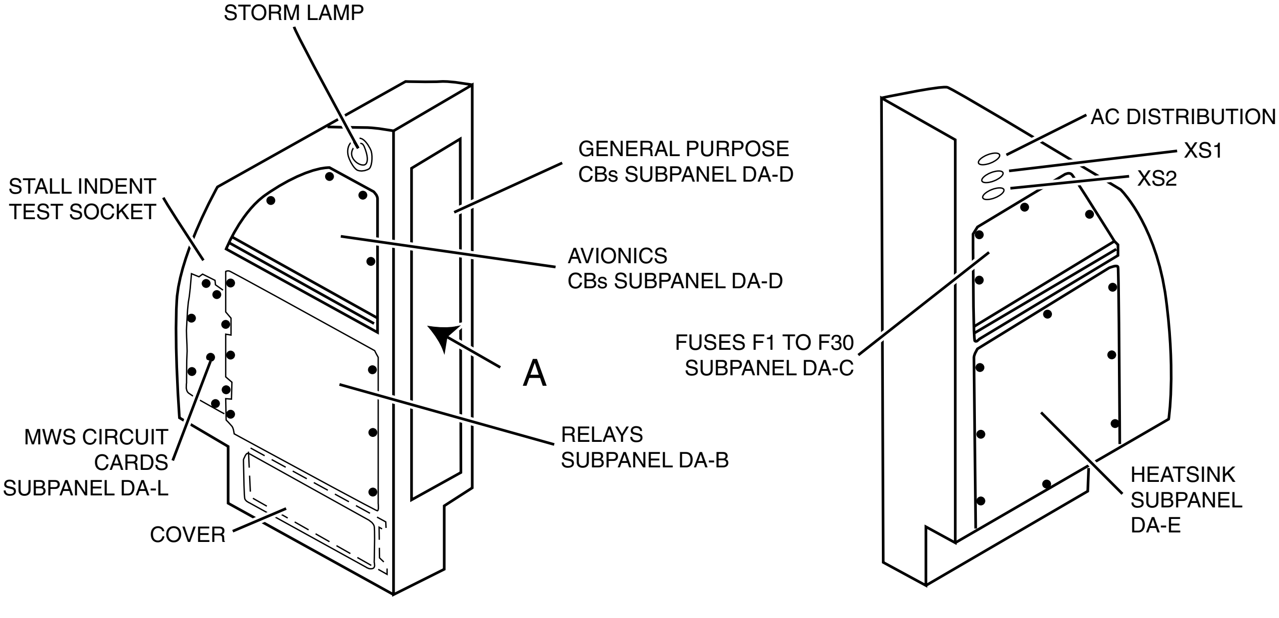


Figure 2-2. Circuit-Breaker Panel Locations


STORM LAMP
STALL INDENT TEST SOCKET
GENERAL PURPOSE CBs SUBPANEL DA-D
AVIONICS CBs SUBPANEL DA-D
A
MWS CIRCUIT CARDS SUBPANEL DA-L
COVER
RELAYS SUBPANEL DA-B
AC DISTRIBUTION
XS1
XS2
FUSES F1 TO F30 SUBPANEL DA-C
HEATSINK SUBPANEL DA-E

LOOKING AFT **LOOKING FORWARD**

Figure 2-3. Main Distribution Panel

DC POWER

BATTERIES

Two types of batteries may be used:

- 24 VDC, 23 ampere-hour nickel-cadium

or

- 24 VDC, 25 ampere-hour lead-acid.

Located in the rear equipment bay(Figure 2-4). There are separate emergency power packs (batteries) for avionics and standby instruments. Their ratings depend on avionics and instrument requirements. For the main batteries, high-temperature warning annunciators are provided in the flight compartment.

The avionics emergency battery powers selected avionics equipment during a total DC power failure. The equipment selected varies between completion centers; however, all installations supply adequate emergency power to the avionic equipment needed for safe landing.

The main batteries maintain DC power to the PE bus bar in the event all generated power fails. The main batteries are also used as an internal power source for engine starting. Start control circuits are supplied from the main batteries via the PE bus bar (See Figure 2-5).

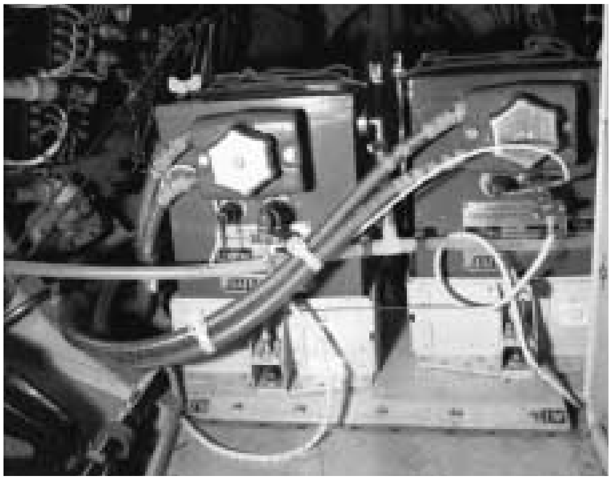


Figure 2-4. Battery Locations

Electrical controls, indicators, and component locations are shown in Table 2-1.

Aircraft with lead-acid batteries do not have the following annunciators:

- BATT HOT
- BATT OVHT

Table 2-1. ELECTRICAL COMPONENT LISTING

ITEM	LOCATION
BATTERY OFF/ON/EMERG switch BATTery ISOLATE switch EXT POWER ON/OFF switch BATT 1 CNTCTR annunciator BATT 2 CNTCTR annunciator EXTernal BATTery CHG ON/OFF switch BATT HOT annunciator BATT O/HEAT annunciator GEN (1 or 2) FAIL annunciator BUS TIE OPEN annunciator Voltmeter Ammeter BAIT AMPS and 0-40 pushbuttons Voltmeter selector switch	Electrical section of roof panel
START PWR annunciator switch STARTER annunciator switches, ENG 1 and ENG 2	ENGINE START section of roof panel
Battery temperature indicator	Center pedestal

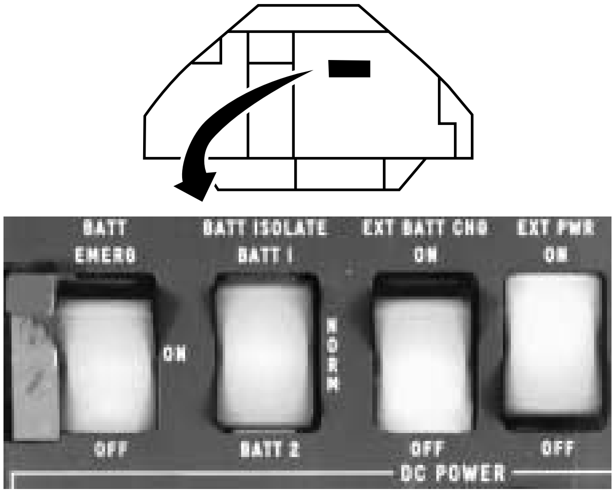


Figure 2-5. DC Bus Bar Control

The three-position BATT OFF/ON/EMERG switch enables Nos. 1 and 2 batteries to be used in the normal (on) or emergency modes. The EMERG position of the switch is protected by a spring-loaded balk.

Switch To ON

With both generators off line and external power selected to OFF, setting the BATT switch to ON uses No. 1 battery to (refer to Figure 2-9) close emergency contactors Nos. 1 and 2. With these contactors closed, No. 1 and No. 2 batteries are connected in parallel to energize bus bar PE via diodes.

Bus bars PS1 and PS2 are not energized unless external power is connected or a generator (main or APU) is on line. When a generator comes on line, both emergency contactors remain closed, and a battery is connected to the bus bar via closure of the appropriate battery contactor. When the buses are tied, both battery contactors close to connect No. 1 and No. 2 batteries to the PS bus bars for battery charging.

CAUTION

No. 1 BATTERY may be excessively discharged and damaged if the roof switch on panel DA and the entry light switch on the forward cabin bulkhead are left ON for long periods while external power is applied.

This is because the vestibule roof light and the entry light are connected via their switches directly to the No. 1 battery and will come on irrespective of the position of the EXT PWR, EXT BATT CHG or BATT switches.

Switch To EMERGency

Setting the switch to EMERG closes both emergency contactors via individual circuits to connect No. 1 and No. 2 batteries to the PE bus bar. The EMERG position also causes the battery contactors to open, thereby disconnecting the batteries from their charging circuit.

When the batteries are fully charged and the correct emergency procedure is carried out, No. 1 and No. 2 batteries can maintain essential services for approximately 37 minutes. If the batteries are less than fully charged and prompt action in off-loading all nonessential services is not accomplished, the time of service is shortened.

BATTery ISOLATE Switch

Setting the switch to BATT 1 or BATT 2 opens the associated battery contactor, isolating the battery from the main bus bar. The associated BATT CNTCTR annunciator on the roof panel illuminates. Setting the switch to NORM permits No. 1 and No. 2 battery contactors to close when the bus bars are powered from the airplane generators or when charging No. 1 and No. 2 batteries from an external power supply.

START PWR Switch

This sequential push-on, push-off switch, with an integral PWR ON/PUSH FOR ABORT annunciator, functions as a start master switch for either external or internal starts. Pressing the START PWR switchlight illuminates the PWR ON/PUSH FOR ABORT annunciator. Starting is now available via STARTER ENG 1 or ENG 2 switch.

During external starts, with start power selected and ground power connected, an interlock with a ground power contactor prevents operation of an internal start contactor.

Internal starting should be used when external power is not available. In these conditions, pressing the START PWR switch energizes an internal start contactor to connect No. 1 and No. 2 batteries to a start bus bar. Bus bar PE remains energized, with the BATT switch on, to supply essential services such as fire protection during engine starting.

Battery High-Temperature Warnings (Nicad Batteries Only)

An amber BATT HOT annunciator and a red BATT OVERHEAT annunciator are connected to thermal switches integral to the No. 1 and No. 2 batteries. Both annunciators are included in the master warning system. (See Appendix B.)

BATT HOT illuminates when the temperature of either battery reaches approximately 57° C, and BATT OVERHEAT warning occurs at approximately 70° C.

Battery Voltage Indication

A voltmeter and selector switch are located on the DC POWER section of the overhead panel. When the selector switch is set to B1 or B2, the voltage of the associated battery is indicated. A third position is provided and, a toggle switch will allow voltage of a third and fourth emergency avionics battery to be read when selected.

Battery Charging

Under normal conditions, with the bus tie closed, the main batteries are charged whenever a generator is on line. Following appropriate manual selection, the rates of charge are displayed on the generator ammeters.

With an external power supply connected and the EXT PWR switch on, Nos. 1 and 2 batteries may be charged by turning the EXT BATT CHG switch to ON. This action energizes both battery contactors (via the ground supply contactor, a battery hot relay and an undervolts/overvolts relay) to connect No. 1 and

No. 2 batteries to PS1 and PS2 bus bars. When the indicated rates of charge reduce to less than 5 amps, EXT BATT CHG switch should be positioned to OFF.

With PE bus bar energized, the relevant BATT 1 or 2 CNTCTR annunciator illuminates whenever the associated No. 1 or No. 2 battery contactor or emergency contactor is deenergized.

NOTE

> If, during external charging, a battery becomes hot or ground supply voltages become abnormal, Nos. 1 and 2 batteries are automatically isolated from the airplane bus bars, and battery charging ceases.
>
> If battery isolation is due to battery overheat, charging automatically resumes when battery temperature drops to a safe level.
>
> In the event of battery isolation due to abnormal ground supply voltages, turn the EXT PWR switch off, disconnect the ground supply, investigate, and rectify. Position the BUS TIE switch to CLOSE before attempting to reinstate.

Battery Temperature Indication (Nicad Batteries Only)

A combined outside air, fuel, battery 1 and 2 indicator is located on the central control pedestal. Temperature sensors fitted integral with No. 1 and No. 2 batteries are electrically connected to the indicator via a rotary selector switch and a push-button switch (Figure 2-6). The relevant battery temperature is indicated when the pushbutton is pressed and the adjacent switch is set to BAT 1 or 2.

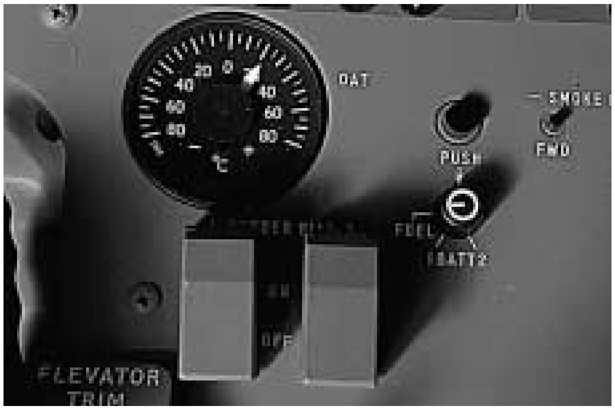


Figure 2-6. Battery Temperature Indicator

GENERATORS

A 30-volt DC, 400-ampere starter-generator is mounted on each engine. The generators are self-exciting and produce rated power output when the engine reaches self-sustaining speed.

Each generator is cooled from an air intake in the engine cowling (Figure 2-7). The cooling air is vented overboard via a duct in the lower cowling. Cooling during ground operation is provided by an integral fan on each generator.

The output voltage of each generator is stabilized by an associated GCU. The GCUs provide load equalization when the generator outputs are tied in parallel.

For engine starting, the generator operates as a starter motor powered by a 28-VDC ground supply or by the airplane main batteries connected in parallel.

The generator controls, indicators, and annunciators, with the exception of an ELECT annunciator on the center instrument panel, are all located on the roof panel.

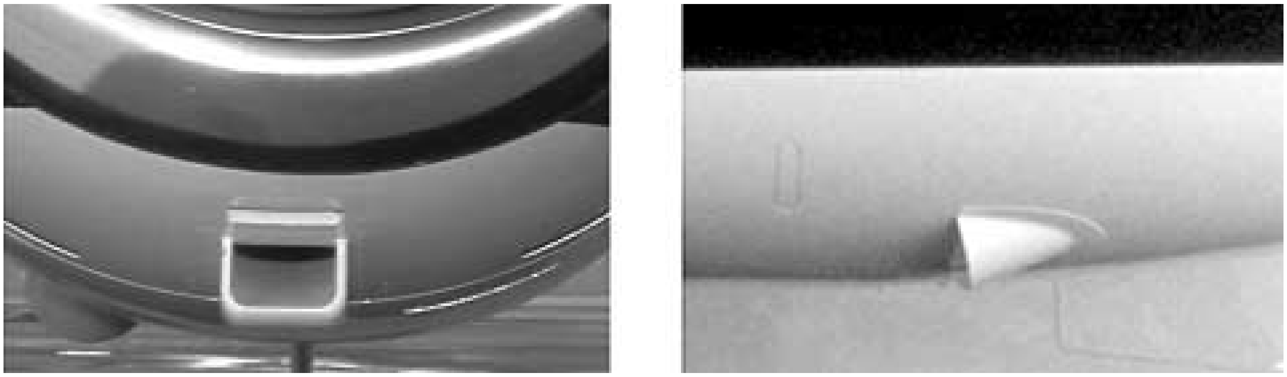


Figure 2-7. Generator Cooling

Engine Starting

The function of the engine starting circuits during the full starting sequence is described, together with the associated controls and indicators, in Chapter 7, "Powerplant." The electrical functions of starting, as well as other circuit configurations, are as follows and are shown in Figures 2-8 through 2-16.

External Start

With external power connected, a supply is available at the EXT PWR switch. Setting this switch to ON completes a circuit, via normally closed contacts of an internal start contactor to energize a ground supply contactor. With this contactor energized, ground power is connected to all bus bars. With the BATT switch at ON position, pushing the START PWR switch on illuminates an integral PWR ON/PUSH FOR ABORT annunciator and brings the start circuit to a state of readiness for ENGINE START switch selection.

Internal Start

Setting the BATT switch to ON and pushing the START PWR switch on energizes the internal start contactor via closed contacts of the deenergized ground supply contactor. These selections connect Nos. 1 and 2 batteries in parallel to the starter bus bar and bring the start circuit to a state of readiness for ENGINE START switch selection.

The PE bus bar remains energized during a normal internal start.

For either type of start the starting sequence may be terminated at any time by a second application of the START PWR switch (now annunciating "PWR ON/PUSH FOR ABORT"). This action deenergizes the internal start contactor and the engine start contactor to isolate the start circuits and extinguish the PWR ON/PUSH FOR ABORT and OPERATING lights.

Power Generation

The transition from starting to generating mode is performed automatically by the GCU. The GCU senses a reference voltage and signals a generator line contactor (GLC) to close, thus connecting the generator to the associated bus bar (PS1 or PS2). When a GLC closes, the associated GEN FAIL light goes out.

In the generating mode, the GCU operates as a voltage regulator. The GCU also signals the GLC to open when reverse current, overexcitation, or overvoltage is detected. Additionally, to avoid reverse current flow, the GCU prevents closure of the GLC when bus bar voltage is higher than the generator terminal voltage.

NOTE

> Overvoltage and overexcitation, in addition to opening the GLC, also trips the generator field circuit. The opening of the field circuit occurs upon detection of these faults whether or not the GLC is closed.

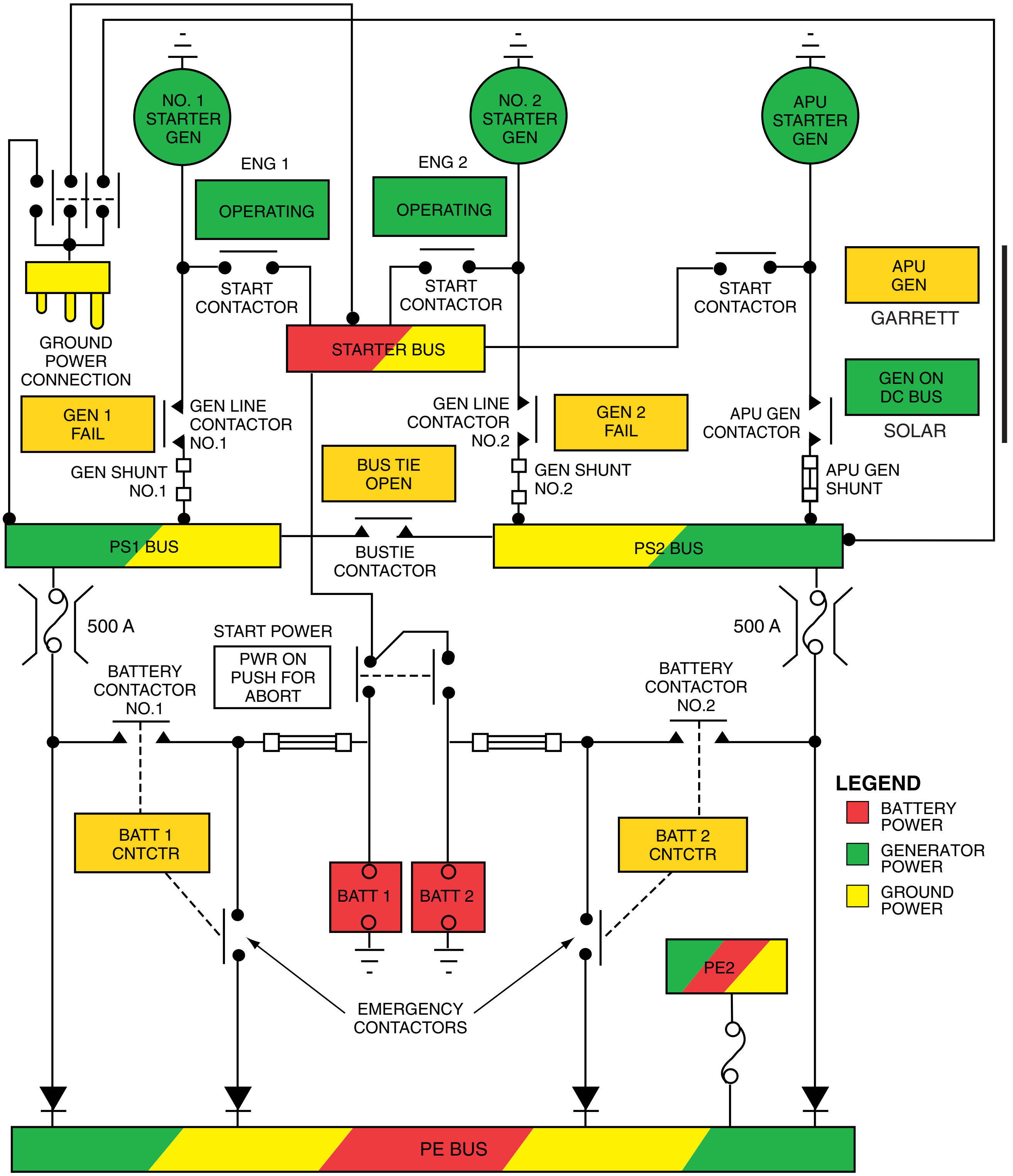


Figure 2-8. Power Generation (Configuration No.1)

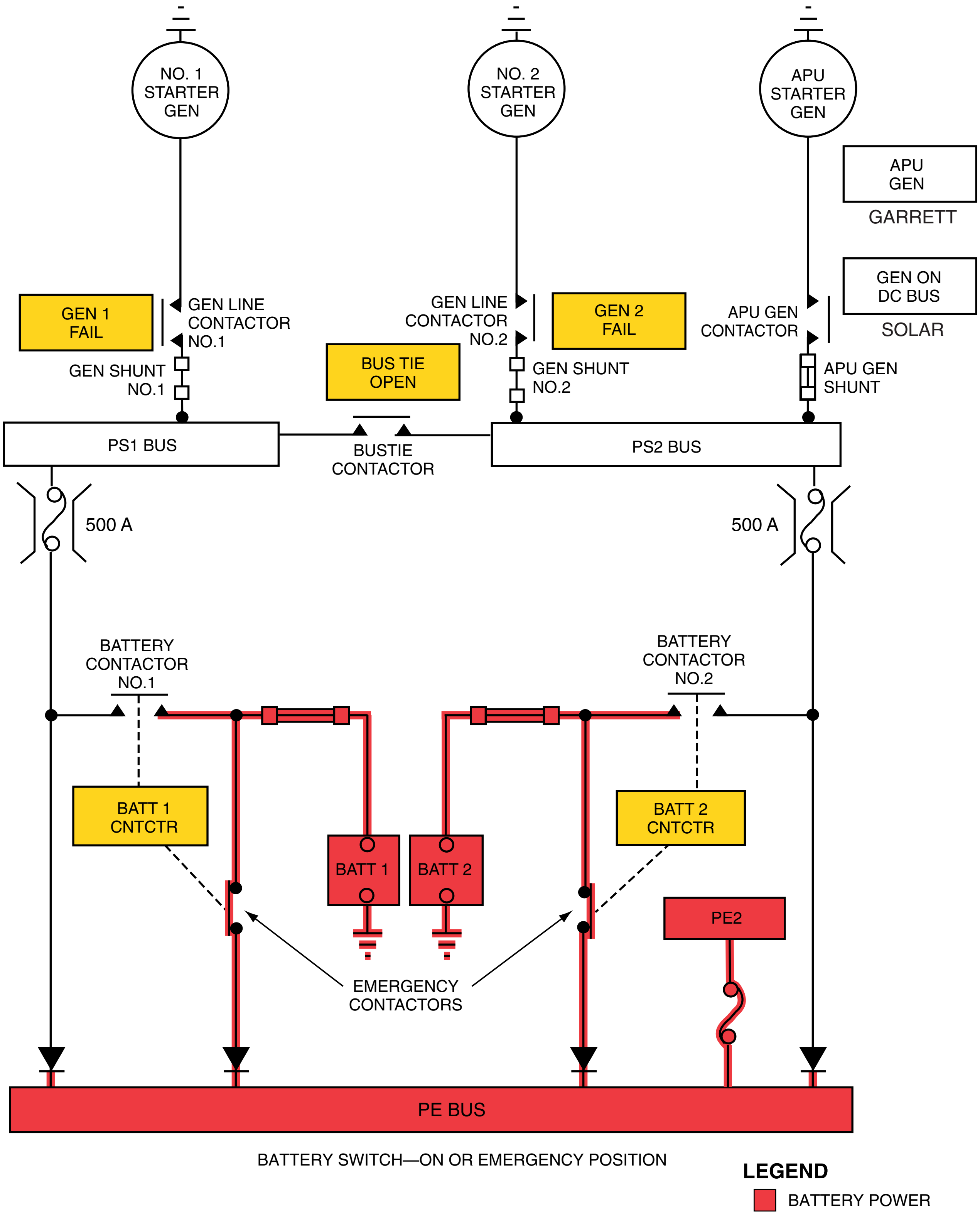


Figure 2-9. Power Generation (Configuration No. 2)

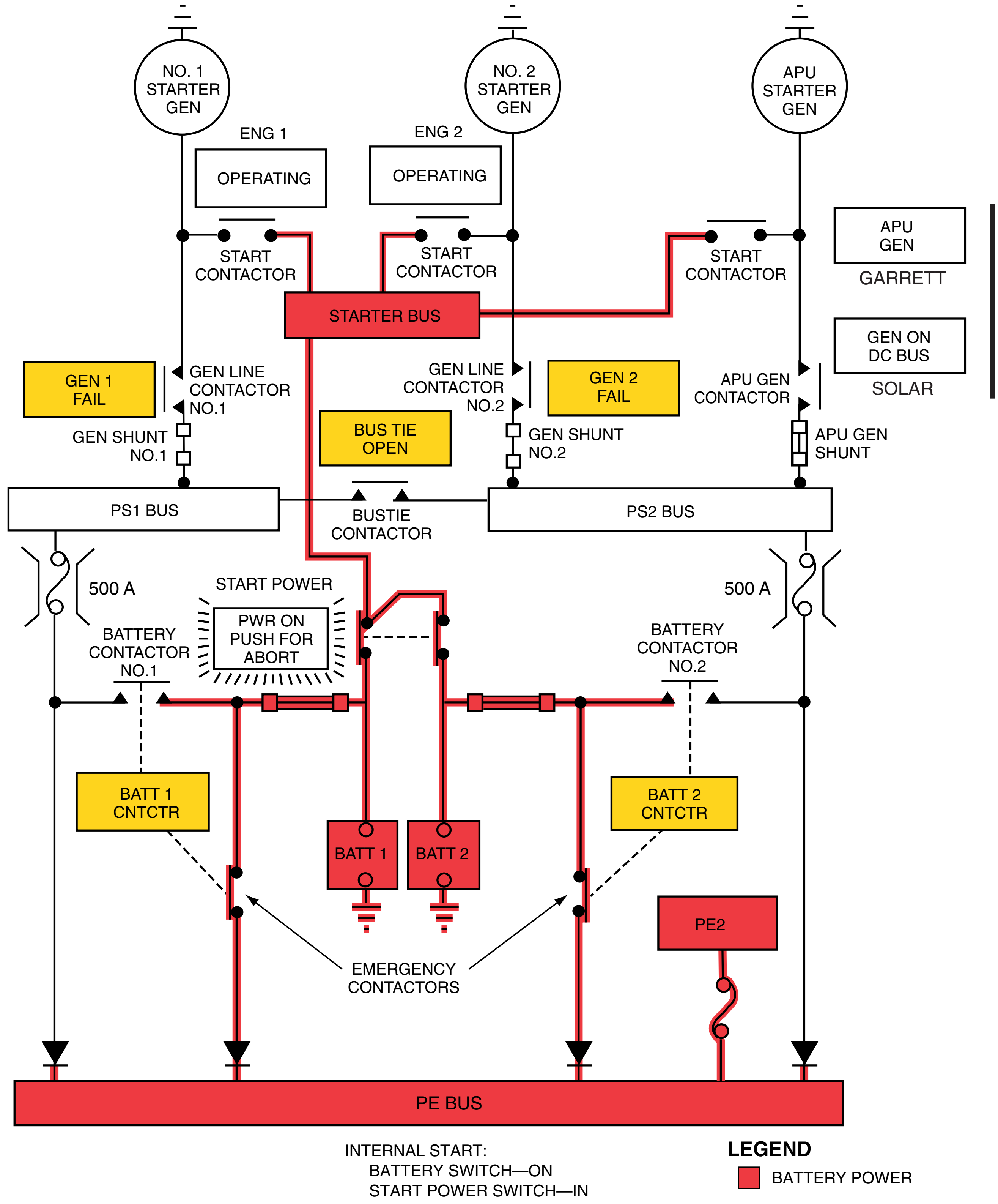


Figure 2-10. Power Generation (Configuration No. 3)

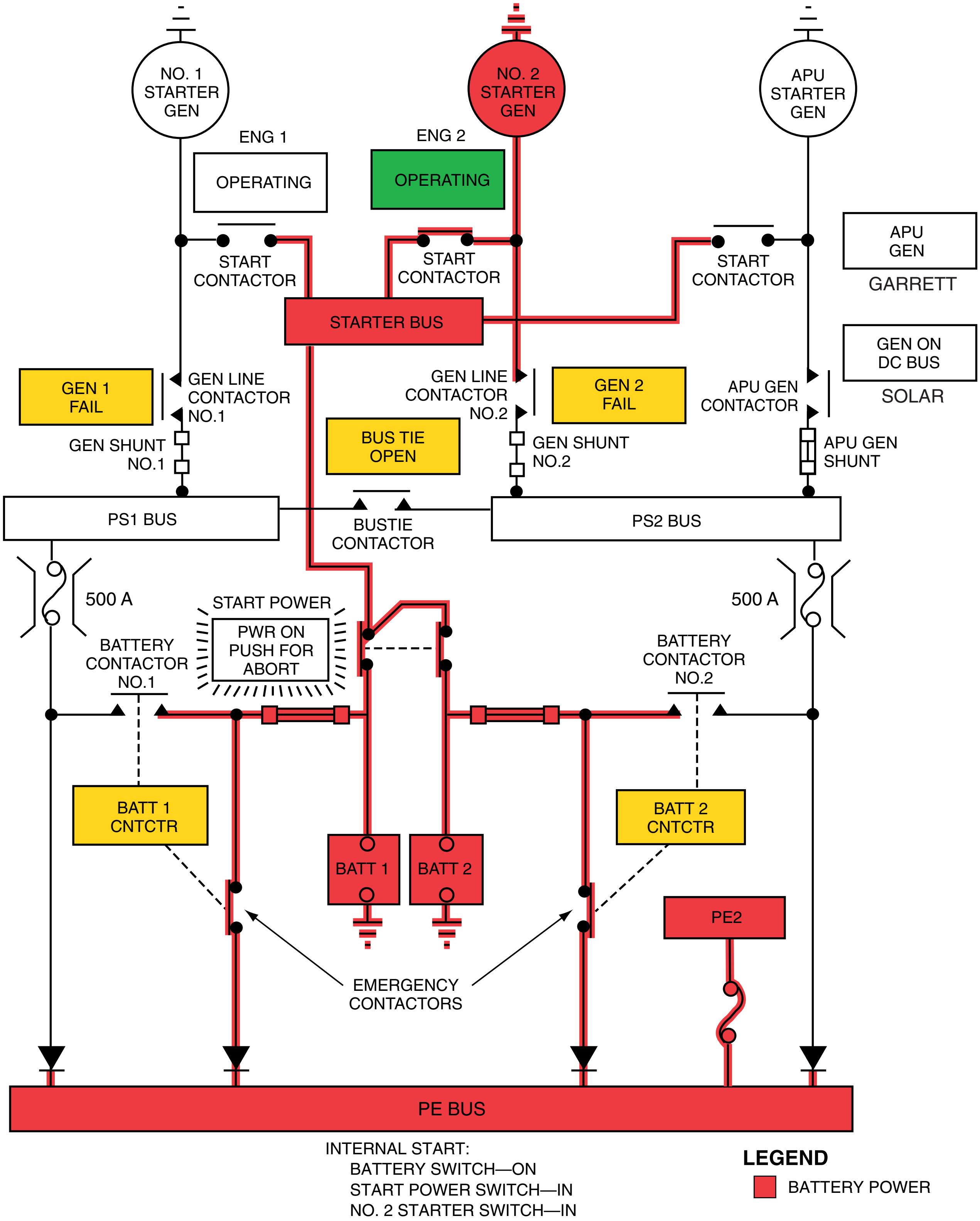
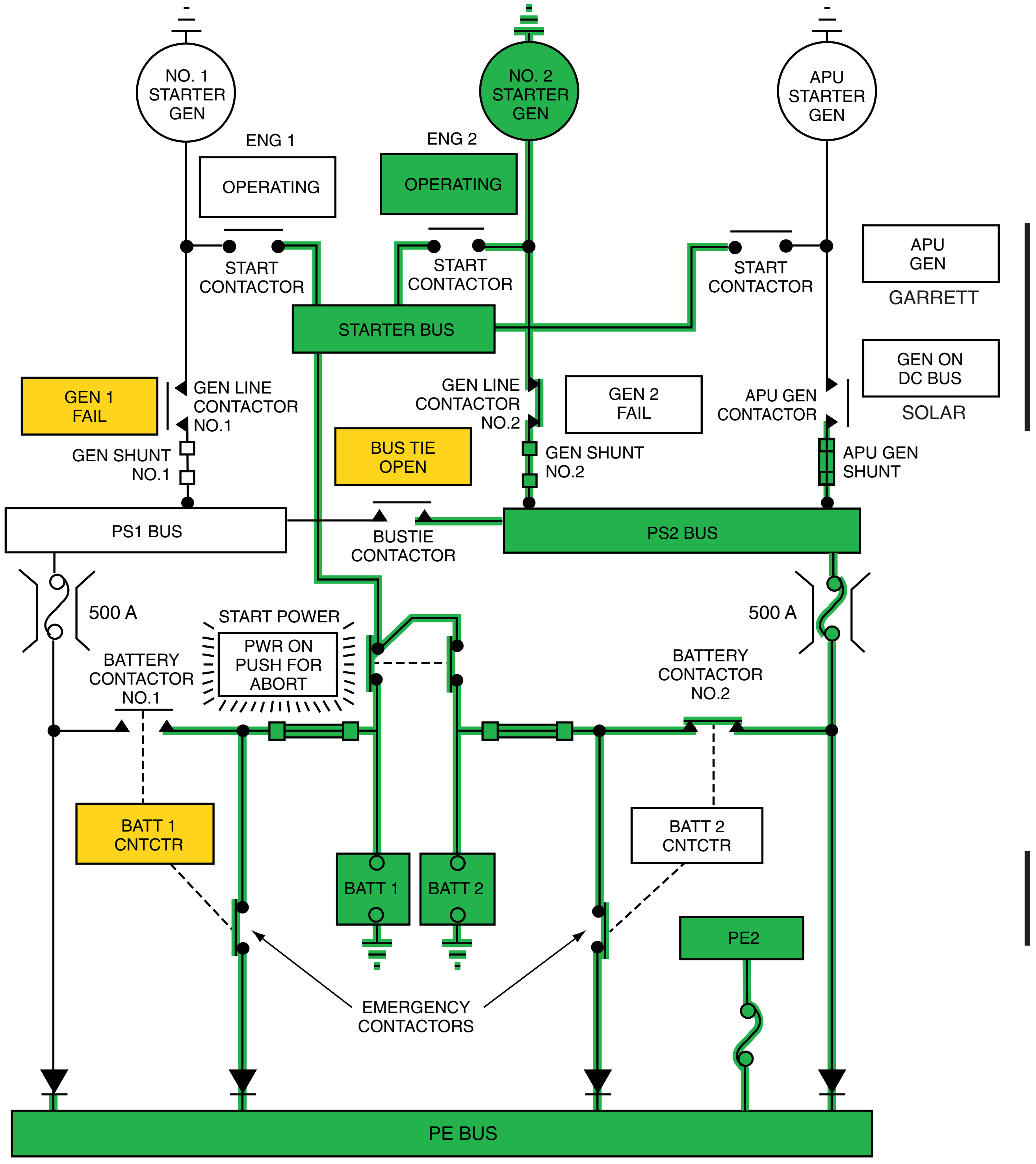


Figure 2-11. Power Generation (Configuration No. 4)

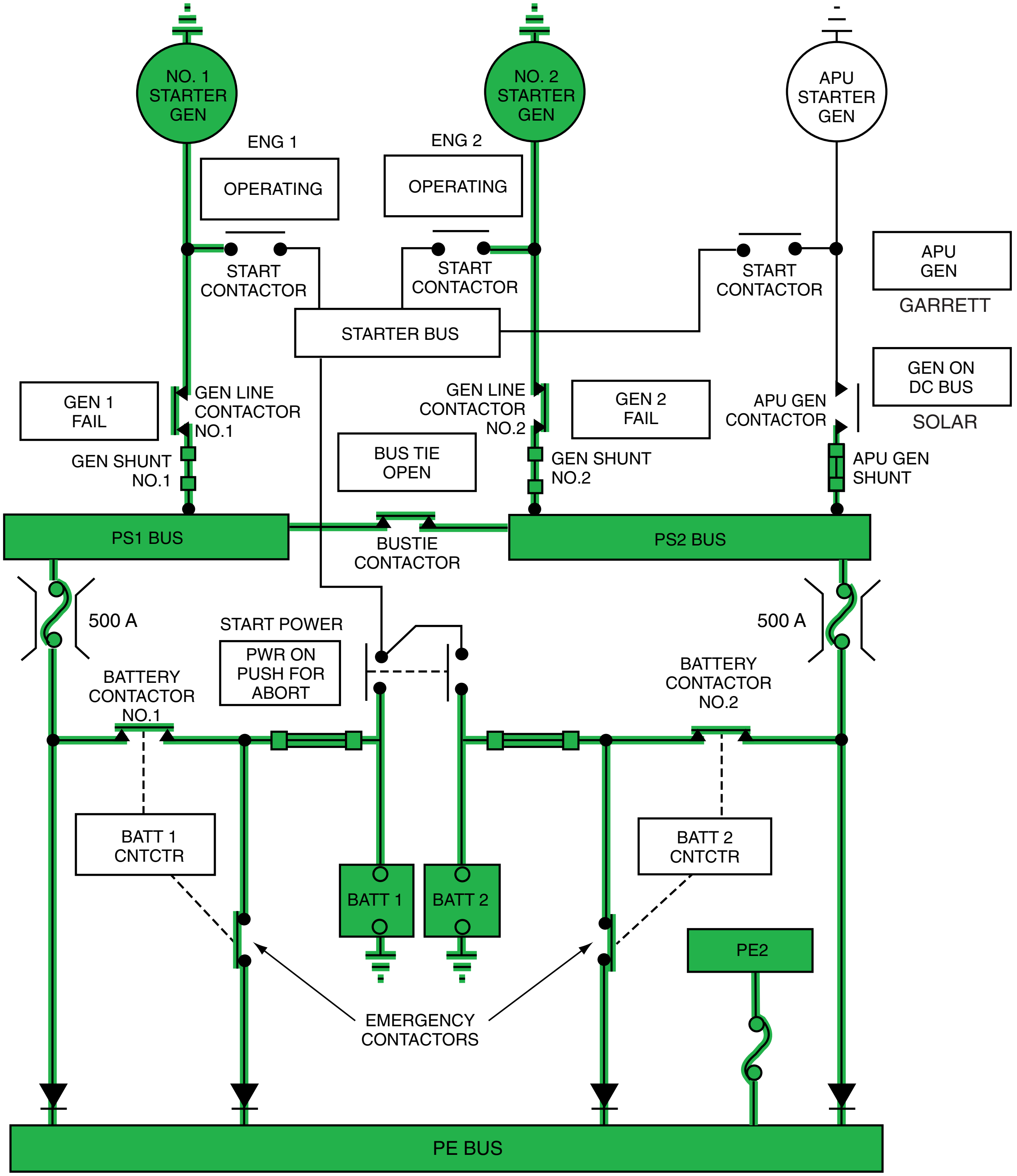


NO. 2 GENERATOR ON LINE
BATTERY SWITCH—ON
START POWER SWITCH—IN
BUS TIE CONTACTOR—OPEN

LEGEND

🟥 BATTERY
🟩 GENERATOR POWER

Figure 2-12. Power Generation (Configuration No. 5)



NORMAL FLIGHT CONDITION:
BATTERY SWITCH—ON
BOTH GENERATORS ON LINE—IN
BUS TIE CONTACTOR—CLOSED

LEGEND

🟥 BATTERY
🟩 GENERATOR POWER

Figure 2-13. Power Generation (Configuration No. 6)

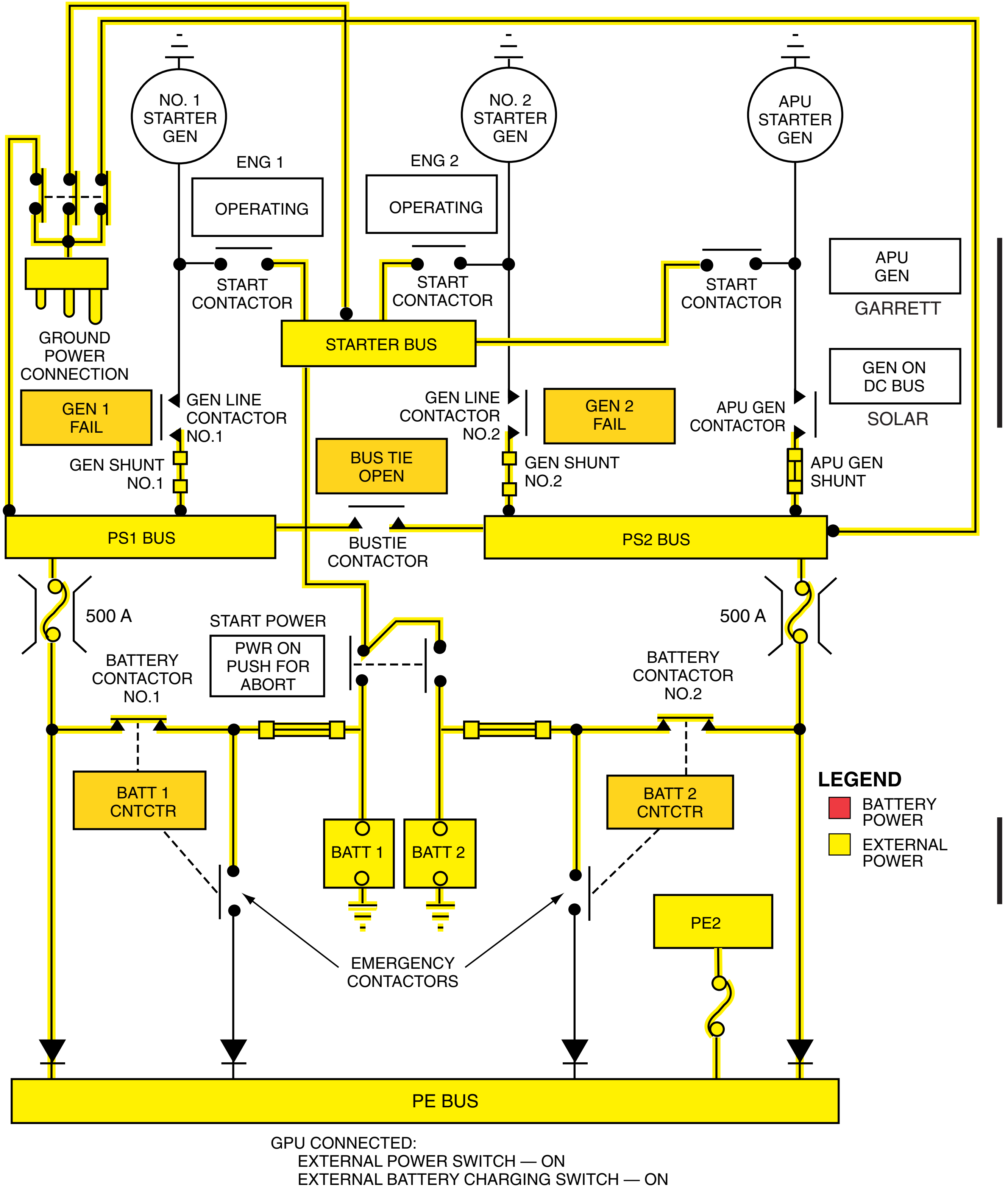


Figure 2-14. Power Generation (Configuration No. 7)

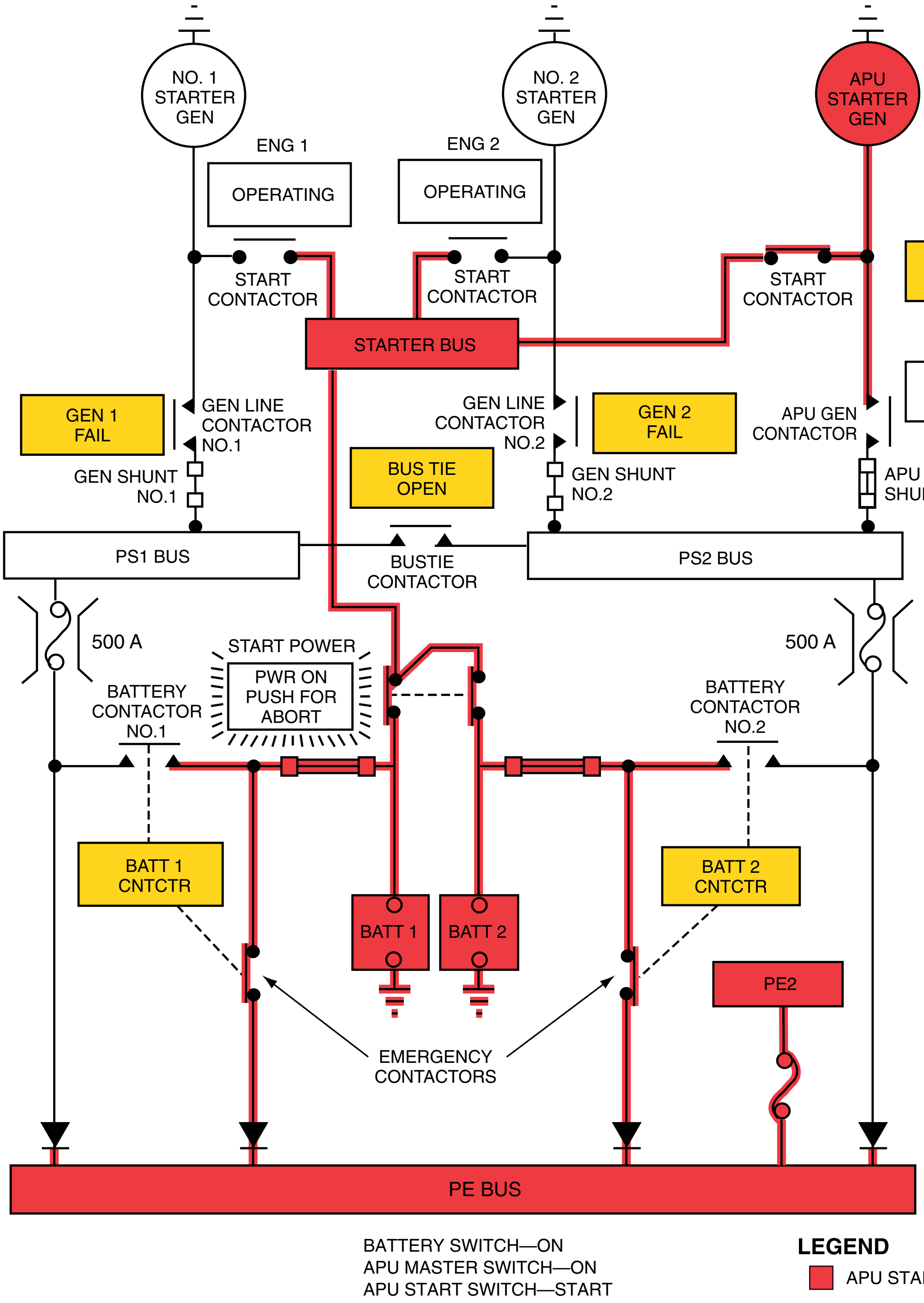


Figure 2-15. Power Generation (Configuration No. 8)

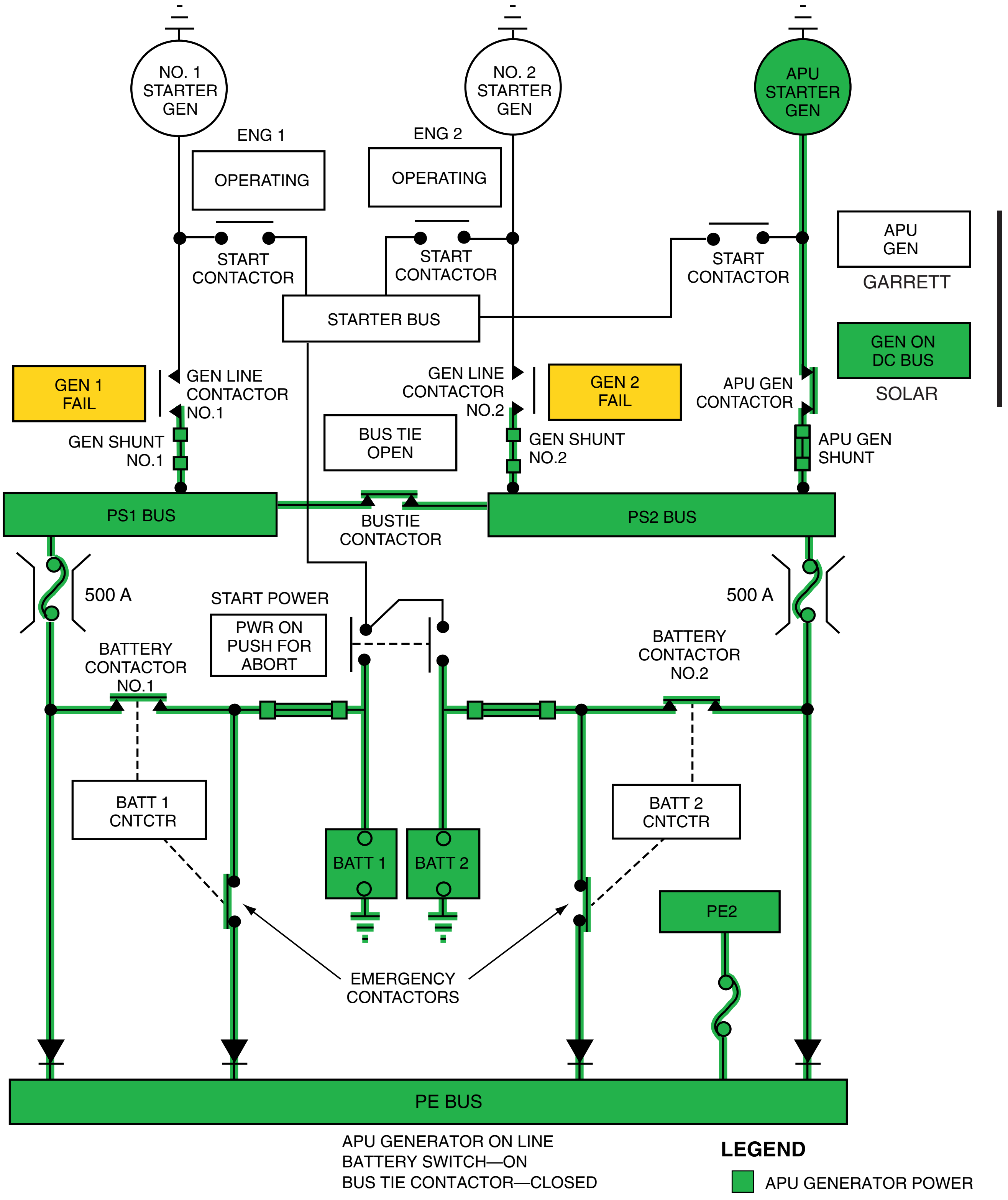


Figure 2-16. Power Generation (Configuration No. 9)

Any fault detection and opening of the GLC takes the generator off line and lights the GEN FAIL and MWS ELECT annunciators. An attempt to reinstate can be made by holding the associated GEN CLOSE/TRIP switch to CLOSE for five seconds and then release.

NOTE

With external power on the aircraft, closure of the GLC is inhibited.

Bus-Tie Contactor

The bus-tie contactor splits the main bus bars in the event of power supply failure or malfunction. The mechanically latched contactor is manually controlled by a spring-loaded BUS TIE CLOSE/OPEN switch or automatically opened by the voltage-sensing unit in overvoltage or undervoltage conditions.

With the bus-tie contactor open, bus bars PS1 and PS2 are split, and the generator-equalizing circuit is open. Additionally, a supply is connected to light an amber BUS TIE OPEN annunciator and ELECT annunciator on the MWS panel. With any airplane generator connected to the bus-bar system, the "close" coil can be operated by positioning the BUS TIE switch to CLOSE. This selection connects a supply from bus bar PE via the generator line contactor to close the bus-tie contactor.

With the bus-tie contactor closed, the BUS TIE OPEN light goes out, the main bus bars are linked together, and the generator-equalizing circuit is completed.

The equalizing circuit ensures equal load sharing between generators when the systems are paralleled and serviceable.

The voltage-sensing unit opens the bus-tie contactor automatically when an overvoltage or undervoltage condition occurs on the main bus bar. The unit incorporates sensing circuits which close a pair of contacts when the voltage exceeds 29.5 ±0.5 volts or falls below 24.5 ±0.5 volts. Time-delay circuits in the voltage-sensing unit prevent the bus-tie contactor from being operated by transients.

With PS1 and PS2 bus bars split, the serviceable channel restores to 28 volts. The unserviceable channel may be switched off automatically by the operation of the GCU overvoltage or reverse-current protection devices. Alternatively, the unserviceable channel can be switched off manually by positioning the associated GENerator control switch to TRIP. With the unserviceable generator off line, positioning the BUS TIE switch to CLOSE closes the bus-tie contactor.

Overvoltage Protection

The GCU overvoltage protection device operates to open the generator field circuit when generator terminal voltage is 32.5 volts or above. At the same time, the GLC is signaled to open and thus disconnect the generator from the associated bus bar (PS1 or PS2).

Generator Control Switches

A three-position GENerator control switch, labeled "CLOSE/TRIP," is connected into each generator circuit. (See Figure 2-17.) The switch, spring-loaded to the center position, can be

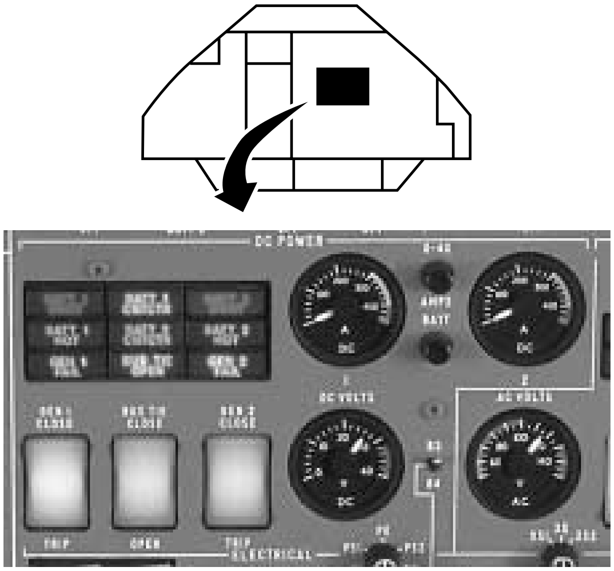


Figure 2-17. Generator Controls and Indicators

used either to reinstate an off-line generator or to take a malfunctioning generator off line.

Momentary positioning of the GEN switch to TRIP opens the GLC and generator field circuit. With a generator off line due to TRIP selection or because of operation of any of the GCU protection devices, holding the GEN switch to CLOSE for five seconds may reinstate the generator. The generator comes back on line, provided no fault exists, when the switch is released.

Ammeters

A dual ammeter is provided (see Figure 2-17). The left ammeter indicates No. 1 generator load or No. 1 battery charge rate; the other indicates No. 2 generator load or No. 2 battery charge rate. A BATT pushbutton and a 0-40 pushbutton are also fitted.

With generators on line, each ammeter normally indicates the load on the associated generator in the range of from 0-450 amperes. Readings should be approximately equal if the equalizing circuit is functioning correctly. The maximum permissible discrepancy between readings is 40amperes.

When the BATT button is pressed, the charge rates of No. 1 and No. 2 batteries are indicated. If the battery charge rate cannot be read easily on the full instrument scale, the 0-40 switch should also be pressed. With both buttons pressed, the value of the graduations on both ammeters is divided by 10, and the ammeter scales represent a range from 0-40 amperes. This facility may only be used when the charging rate is less than 40 amperes. (Immediately after an internal start, the charging rate may be as high as 150 amperes.)

A red segment is marked on each instrument dial to cover the range above 400 amperes; the 300-400 ampere range is marked in orange.

Voltmeter

A voltmeter (Figure 2-13) indicates the voltage of the installed batteries or the voltage on bus bars PS2, PE, or PS1 when an adjacent six-position selector switch is set appropriately. The positions are labeled “B1-B2-PS1-PE-PS2-AUX 3-4.”

APU

Either of two types of APUs may be installed (Figure 2-18). Both perform the same functions, which are to provide a supply of DC electrical power and bleed air for air conditioning on the ground. The two manufacturers are Garrett/AiResearch and Solar. While control configuration

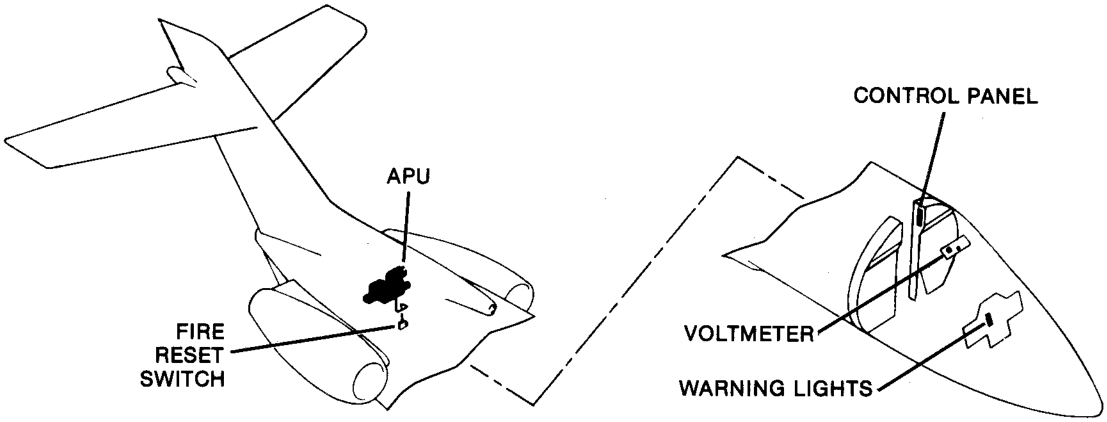


Figure 2-18. APU Location

varies slightly, their general operation is quite similar.

GROUND POWER

A standard 3-pin, 28-volt DC ground power receptacle is provided at the right rear fuselage (see Figure 2-19).

For external starts, a ground power unit capable of producing 28 volts DC with a minimum output of 42 KW (short term capability of 28 volts at 1,500 amperes) is required. The ground power unit must be fitted with a current limiter whose operating rating is 1,100 amperes.

For ground testing the airplane electrical services, a 28-volt DC battery cart is adequate.

The EXT PWR ON/OFF switch (Figure 2-20) is located in the electrical section of the roof panel.

Ground power supplies are connected to the airplane bus-bar system and to the starter supply lines through a ground supply contactor. Contactor operation is controlled by an EXT PWR ON/OFF switch supplied from the third pin of the ground supply plug.

Switching EXT PWR on (refer to Figure 2-20) connects the ground power source to the bus bars. In this condition, auxiliary contacts in the ground contactor normally ensure that the airplane batteries and/or generators are isolated from the bus bars. However, manual selection of an external battery-charging switch connects Nos. 1 and 2 airplane batteries to the bus bars, thereby allowing them to be charged from a ground power supply.

With EXT PWR switched on and EXT BATT CHG switched off, the airplane batteries are isolated from the bus bars, and positioning of the BATT switch to ON has no effect except when attempting to start the engines or APU.

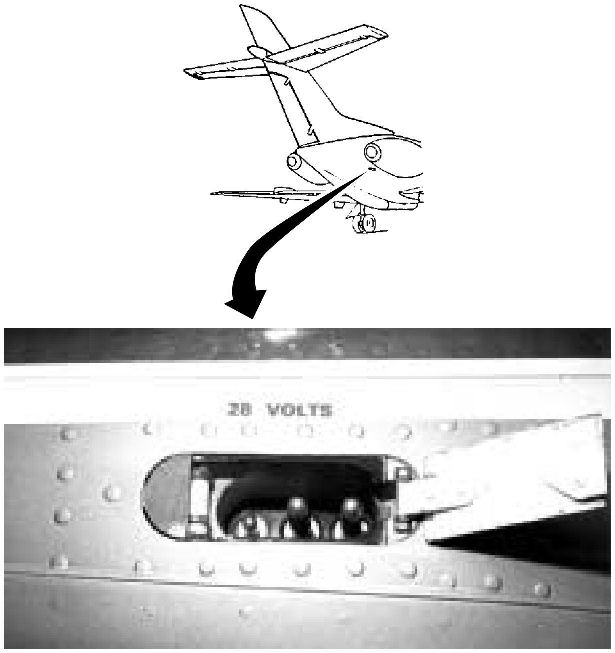


Figure 2-19. Ground Power Receptacle

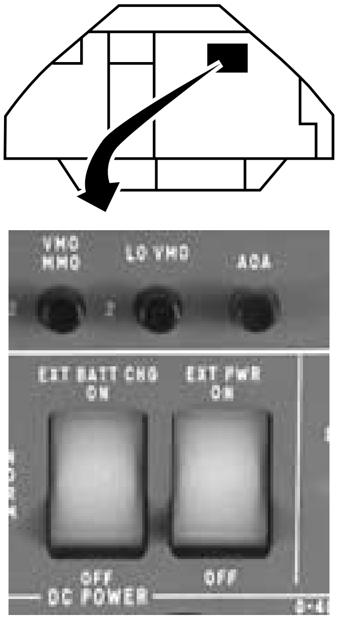


Figure 2-20. Ground Power Switch

A start inhibit facility prevents engine starting while the EXT BATT CHG switch located on the pilot's roof panel is set to ON.

NOTE

With the BATT switch on and ground power plugged in, selecting START PWR on first causes the EXT PWR switch to be ineffective.

DISTRIBUTION

Distribution of electrical power is made via a main distribution panel DA located just aft of the copilot's seat. The panel embodies two general service circuit-breaker panels (Figures 2-21 and 2-22) and one circuit-breaker panel for avionics, navigation, and radio services.

The general-service circuit breakers on the subpanel DA-D and DA-A are identified by labels showing the systems and a grid. The grid is marked in numbers horizontally and letters vertically. Circuit breakers protecting unswitched PE systems are marked with a white background for ease of isolation in the event of smoke or fire. The push-pull button of each circuit breaker is marked with the appropriate ampere rating.

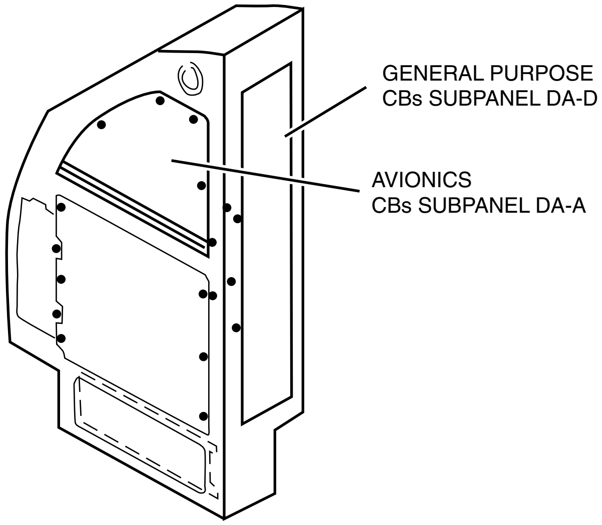


Figure 2-21. Circuit Breaker Panel DA-A

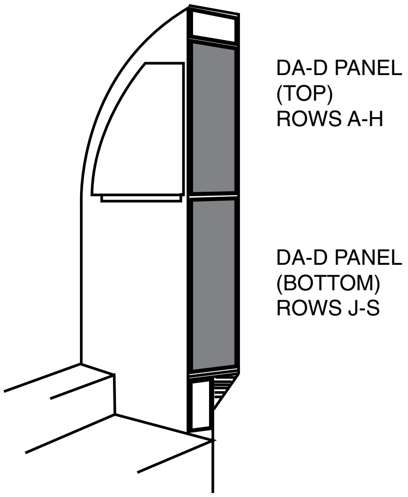


Figure 2-22. Circuit Breaker Panel DA-D

The DC bus bar system also includes three panels in the rear equipment bay area. These panels contain circuits and fuses for certain essential services. A negative ground return is provided through the airplane structure via grounding points below the panels in the equipment bay.

Nonessential systems which are duplicated, or where the load is divisible, are supplied from bus bar PS1 or PS2 so that a power supply failure on either bus bar will not result in a complete failure of these systems. Nonessential systems which are not divisible are supplied from either PS1 or PS2. All essential systems are supplied from bus bar PE.

OPERATION

If a ground power supply is available, check that the plug is in firmly. The ground power unit must be fitted with a current limiter operation at 1,100 amperes.

Before starting the external inspection, carry out a preliminary check in the flight compartment. Switch on the entry light if required and confirm that the fire extinguisher switches are off and that all circuit breakers are in. If ground power is not connected, set the EXT BATT CHG switch to OFF, the EXT PWR switch to OFF, and the BATT switch to ON. With ground power connected, leave the EXT BATT CHG switch off, and set the EXT PWR switch to ON. Press one MWS red lamp to cancel both. Press the MWS TEST button and check that all the MWS annunciators illuminate, the ELECT repeater flashes, and the MWS red lamps flash. Release the test button and cancel the MWS flashers. Switch on minimum flight compartment lighting and check that all switched PE loads, fuel pumps, external lights, pitot heaters, radio, and radar instrument switches, etc., are off. A battery capacity check as found in the AFM should be accomplished.

If ground power or APU power is not available, return the BATT switch to OFF until immediately prior to engine starting.

AC POWER

Power for windshield and side window heating is supplied from two variable-frequency alternators. For information on the windshield anti-icing system, refer to Chapter 10, “Ice and Rain Protection.”

INVERTERS

Single-phase, 115-volt, 400-Hz AC is provided by two static main inverters installed in the rear equipment bay. Normally, the No. 1 inverter feeds XS1 and XE distribution lines, and the No. 2 inverter feeds XS2. Either inverter is, however, capable of feeding all distribution lines, and changeover is achieved automatically if one inverter fails.

In the event of failure of both main inverters, AC supply is automatically provided by a standby static inverter installed in the rear equipment bay. The output from this inverter is connected to distribution line XE and supplies only the essential AC-operated systems.

Both 6- and 26-volt AC supplies are provided via two transformers. The No. 1 transformer is powered from the XE bus and the No. 2 transformer from the XS1 bus.

Inverter controls, indicators, and locations are shown in Figure 2-23 and are contained in Table 2-2.

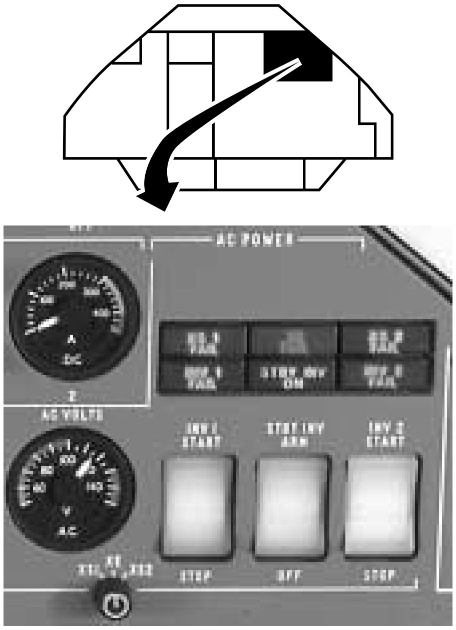


Figure 2-23. Inverter Controls and Indicators

Table 2-2. INVERTER CONTROLS AND LOCATIONS

ITEM	LOCATION
STOP/START switches, INV 1 and 2	ELECTRICAL AC POWER section of the roof panel
STBY INV ARM/OFF switch	
Voltage selector switch, XS1, XE, and XS2	
VOLTS meter	
STBY INV ON annunciator	
FAIL annunciators INV1, INV2, XS1, XS2 (amber), and XE (red)	
ELECT annunciator	MWS center panel

CONTROL AND OPERATION

The AC system is shown schematically in Figure 2-24. Each main inverter is controlled by a three-position START/STOP switch. The switch is spring loaded from START to the center (normal running) position. No. 1 and No. 2 inverters and the distribution lines (XS1, XS2, XE) are each provided with a FAIL annunciator. With the main and standby inverters off, all five annunciators are illuminated. Any AC FAIL warning also activates the MWS flashers.

Holding the INV 1 switch to START energizes a contactor and a relay to start the inverter and connect its output to XS1 distribution line. Contacts in the protection unit close to provide a lock-on circuit for the contactor and to extinguish the INV 1 FAIL light. The INV switch should now be released to the center position. Energizing XS1 operates a relay to extinguish the XS1 FAIL light and connect XS1 with XE. Another relay, operated from XE, extinguishes the XE FAIL light. The XS2 FAIL light is extinguished because XS2 is powered via the relaxed No. 2 inverter relay. Positioning the control switch to STOP deenergizes the contactor and shuts down the inverter.

The No. 2 inverter system operates similarly but supplies XS2 and, when brought on line, extinguishes the INV 2 FAIL and the XS2, XS1, and XE failure lights.

In the event of failure of one main inverter, the associated FAIL annunciator illuminates, the MWS ELECT repeater flashes, and the applicable inverter relay changes over to connect the output of the serviceable inverter to XE, XS1, and XS2 lines; the contactor is deenergized and the failed main inverter shuts off. With the changeover complete, the XS1, XS2, and XE FAIL annunciators of the serviceable inverter remain out.

If any distribution line becomes deenergized, an associated relay opens to illuminate its FAIL annunciator and to operate the MWS ELECT repeater.

The STBY INV switch is set to ARM for flight. With ARM set, the standby inverter immediately comes into operation in the event of failure of both main inverters. The output of the standby inverter is connected only to the XE line. A STBY INV ON annunciator illuminates whenever the standby inverter is operating.

6-Volt and 26-Volt Supplies

Two unswitched transformers convert the 115-VAC supply from the inverters into 6 VAC and 26 VAC. These voltages are required for operation of some of the navigation equipment.

DISTRIBUTION

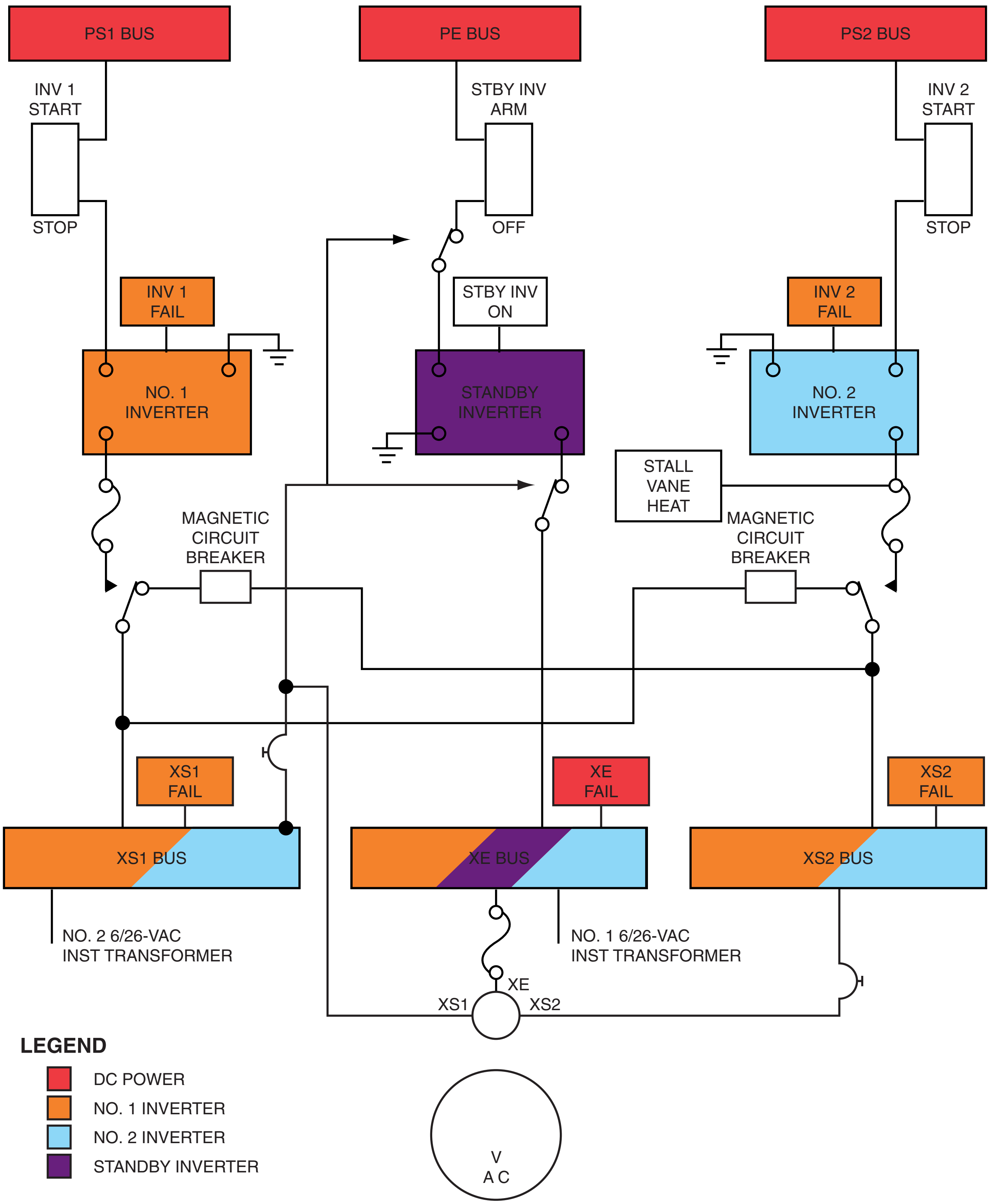
General

Constant-frequency AC outputs from the inverters are fed to certain airplane systems via distribution terminal blocks (bus bars) XS1, XS2, and XE, and relevant circuit breakers. The general service AC circuit breakers are located on a subpanel just aft of the copilot’s seat.

OPERATION

With DC power established, turn the inverters on and check that the MWS lights are extinguished.

Check the inverters as follows: Before starting the No. 1 and No. 2 inverters, switch off the standby inverter and ensure that the XE FAIL annunciator is on. Move the switch to ARM and ensure that the XE FAIL annunciator is extinguished. Start the No. 1 inverter. Check that the associated INV 1 FAIL, XS1 and 2 FAIL and XE FAIL annunciators are extinguished. Start the No. 2 inverter and stop the No.1 inverter, then check that the INV 1 FAIL annunciator comes on. Restart the No. 1 inverter and check that all AC FAIL annunciators are out.



LEGEND

- 🟥 DC POWER
- 🟧 NO. 1 INVERTER
- 🟦 NO. 2 INVERTER
- 🟪 STANDBY INVERTER

Figure 2-24. AC Power System

Both the No. 1 inverter and the No. 2 inverter must be started (INV 1 FAIL and INV 2 FAIL annunciators out) and the standby inverter selected to ARM before takeoff and left in this condition throughout the flight.

ELECTRICAL POWER SOURCES

PE bus bar

- Standby inverter
- Left SCREEN HEAT ON/OFF

PS1 bus bar

No. 1 inverter

PS2 bus bar

No. 2 inverter

XE line

No. 1 26 VAC transformer

XS1 line

No. 2 26 VAC transformer

EMERGENCY BATTERY SUPPLY

No. 3 battery (following a double generator failure) will be automatically applied to the standby horizon to maintain the attitude display and internal lighting on the standby horizon. The E BAT ON annunciator on the bottom half of the standby horizon switchlight capsule will illuminate and an aural warning will sound. The aural warning may be canceled by depressing the horizon warn cancel pushbutton adjacent to the standby horizon.

In addition to the above, the standby altimeter and airspeed indicators will have their internal lighting provided by the No. 3 battery, and the No. 1 N_1 indicator will be powered by the No. 4 battery.

No. 4 battery will provide time limited backup power to the No. 1 AHRS in flight only in the event of a total power failure of the normal aircraft power system.

Radio Emergency Functions

The Radio Emergency System is activated by raising the red RADIO EMERG switch (located at the top of the DA panel behind the copilot) to the ON position. When ON, the Radio Emergency System, No. 4 battery, will provide electrical power to the following systems:

- COMM 1
- NAV 1
- RMU 1
- ATC 1
- CDC

NOTE

Audio is available on headsets only via the emergency position on the respective audio panel.

LIMITATIONS

DC POWER

Batteries

Minimum battery voltage is 23 volts. Maximum battery charge rate prior to takeoff is 20 amps.

Generators

The generators are limited to 300 amperes continuous.

Ground Power

A standard three pin 28 VDC external power receptacle is located in the right rear fuselage.

For external starts, a ground power unit capable of supplying 28 VDC with a minimum output of 42 kW (short term capability of 28 VDC x 1,500 amps) and fitted with a limiter operating at 1,100 amps must be used.

For ground testing the aircraft electrical services, a 28 VDC battery trolley is adequate.

QUESTIONS

1. The electrical rating for each main battery is:

A 23 volts, 34 ampere-hours
B. 24 volts, 23 ampere-hours
C. 30 volts, 24 ampere-hours
D. 28 volts, 23 ampere-hours

2. The batteries will maintain essential service during a dual generator failure for:

A. 10 minutes maximum
B. 20 minutes minimum
C. 37 minutes minimum
D. 150 minutes maximum

3. The maximum battery amps for takeoff is:

A. 25 amperes
B. 30 amperes
C. 35 amperes
D. 20 amperes

4. When a generator is off the line and the GEN FAIL and ELECT annunciators are illuminated, the generator can be reinstated by:

A. Positioning the GEN switch to TRIP, then releasing it
B. Resetting the generator circuit breakers
C. Positioning the GEN switch to CLOSE then releasing it
D. Turning the EXT PWR switch to ON

5. The battery-charging rate can be monitored by:

A. Depressing the BATT ammeter pushbutton
B. Turning off one battery and monitoring the voltmeter
C. Depressing the generator ammeter 0-40 button
D. Monitoring the ammeter with one battery off

6. The ground power receptacle is located:

A. On the left forward fuselage
B. On the left engine nacelle
C. On the right rear fuselage
D. In the right main gear wheel well

7. For engine starting, the ground power unit must produce:

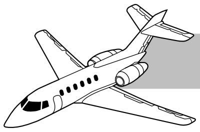
A. 28 volts, 1,000 amperes
B. 28 volts, 1,500 amperes
C. 30 volts, 1,100 amperes
D. 28 volts, 1,200 amperes

8. Circuit breakers protecting unswitched PE circuits are marked to ease isolation in the event of smoke or fire with:

A. Fluorescent paint
B. A white background
C. A large irregular-shaped bead
D. A grid of numbers and letters

9. The electrical power supply for the standby inverter is:

A. PE
B. PE2
C. PS1
D. PS2



CHAPTER 3
LIGHTING

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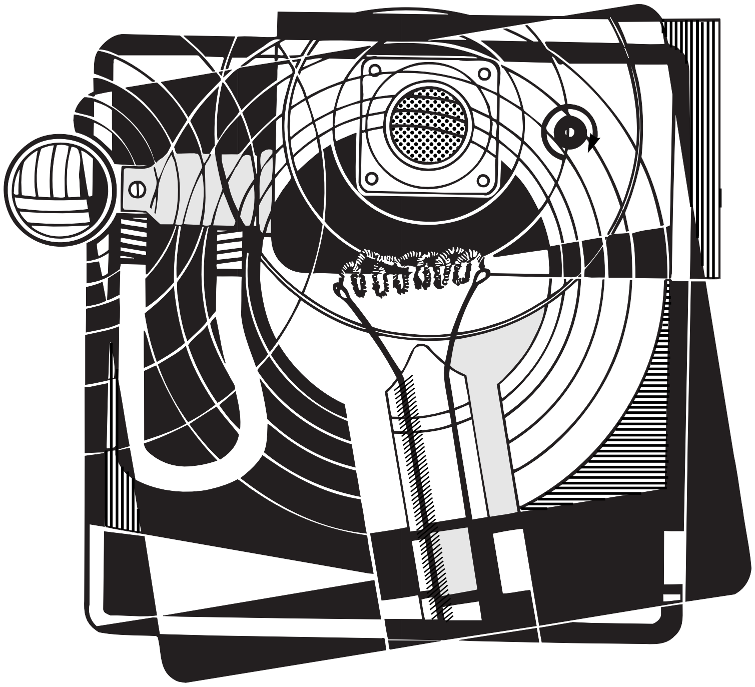
CHAPTER 3
LIGHTING



INTRODUCTION

A standard lighting package is used on the Hawker 800 XP. In the cockpit, lighting is used for general area illumination and for all flight instruments. The majority of the instruments are internally lighted. For direct illumination, floodlights or pillar-mounted lights are provided. The standard warning lights are available for the cabin area, and provisions are made for illuminating the exits in the event of an emergency. General cabin lighting configurations vary greatly due to custom installations. Exterior lighting consists of navigation, landing and taxi, anticollision beacons, strobe, and wing inspection lights.

GENERAL

The main instrument panel lighting is provided by white lights mounted on the lower canopy rail and under the main instrument panel glareshield. The lights are controlled by a dimmer switch on the panel lights section of the left or right glareshields labeled "PANEL LTS-FLOOD." In addition to instrument panel lighting, various instruments and panel-mounted displays are integrally illuminated. In areas where panel or instrument lighting is desirable, and no integral lighting exists, panel-mounted pillar lights are utilized.

Low-voltage power for lighting is obtained from transformers through dimming control units supplied from the DC and AC bus bars. The bus bar supplies are controlled by a push-button LTS MASTER switch on the left side of the roof panel. When pushed on, the switch head illuminates the word "ON," and completes the bus-bar circuit to the dimming control units.

An entry light on-off switch located on the forward bulkhead adjacent to the entry door controls a vestibule light, step light, refuel panel lights, and ankle duct lamps. A roof light switch on the forward bulkhead (DA) panel also controls the lights.

The airplane exterior lighting system is equipped with the following lights:

- Three navigation lights (red on the left wingtip, green on the right wingtip, and white on the tail cone)
- Two flush-mounted landing lights, one located on the leading edge of each wing
- Two taxi lights, one mounted next to each landing light
- Two red anticollision (beacon) lights, one mounted on top of the vertical stabilizer and one on the bottom of the fuselage
- Three strobe lights, one located on each wingtip and one on the tail cone
- Two wing ice inspection lights

INTERIOR LIGHTING

COCKPIT LIGHTING

The cockpit lighting consists of floodlights, edge lighting of panels, and internal lighting of instruments.

Most of the lighting is controlled by switches and dimmers, either on the left or right glareshields (Figure 3-1).

Chart Lights

Two chart lights are mounted on the crew services panel. One, above the copilot's head, is directed toward the pilot's position; the other, above the pilot's head, is directed toward the copilot's position. Normal control of these lights is via dimmer switches mounted on the same panel. An alternate supply is also provided for these lights from the three-position CHART LIGHT/WANDER LIGHT switch (Figure 3-2) on the bulkhead panel facing forward, behind the copilot's seat.

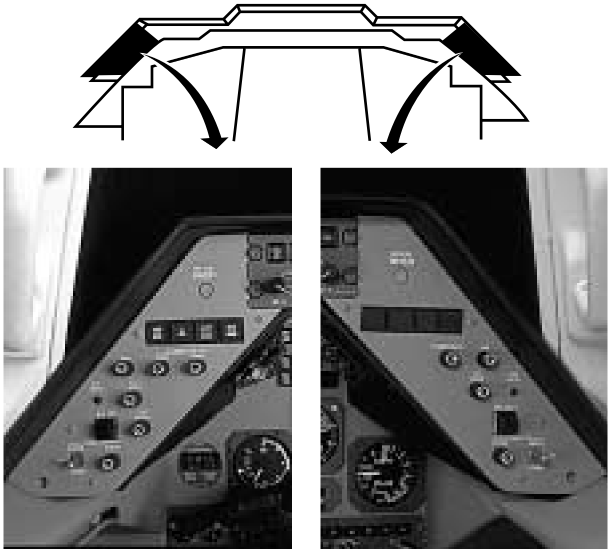


Figure 3-1. Dimmer Switches

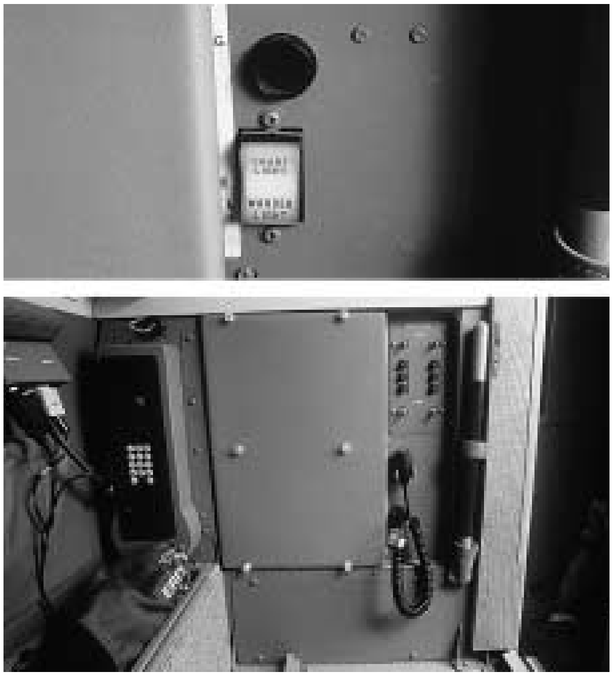


Figure 3-2. CHART LIGHT/WANDER LIGHT Switch

This switch, when positioned to the CHART LIGHT position, connects a power supply via diodes to both chart lights and bypasses the dimmer switches. Both circuits are supplied from the PE bus bar.

Wander Light

The wander light and switch are located adjacent to the aisle behind the copilot's seat. A socket for the light is provided adjacent to the container, and power supplies to the socket are controlled by the three-position CHART LIGHT/WANDER LIGHT switch (Figure 3-2). Supply is from the PE bus bar.

Pad Lights

Two lights, mounted at the tip of the left and right windshield pillars to illuminate the writing pads, are controlled by dimmer switches mounted on the left and right glareshields (Figure 3-3). Supply is from the PS1 bus bar.

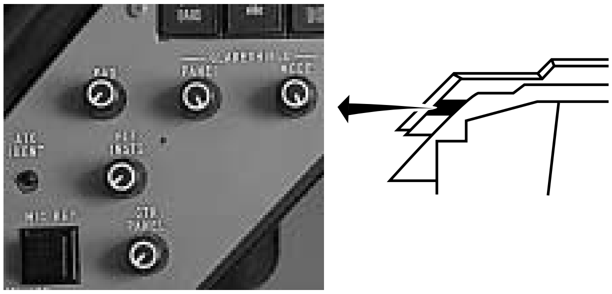


Figure 3-3. Pad Lights Control

Storm Lights

Three lights are provided to give high-intensity white light to prevent crew flash blindness during lightning conditions. The lights, one on the flight compartment right bulkhead and one above each side console, are controlled by two EMERG–OFF–STORM switches on the pilot's and copilot's glareshields (Figure 3-4). Supply is from the PS2 bus bar.

Standby Lighting

Electrical power for standby lighting is available when the PE bus bar is energized. The following lights are connected to the PE bus bar and are controlled by their normal operating switches to provide standby lighting for:

- Chart lights
- Wander light
- Spotlight

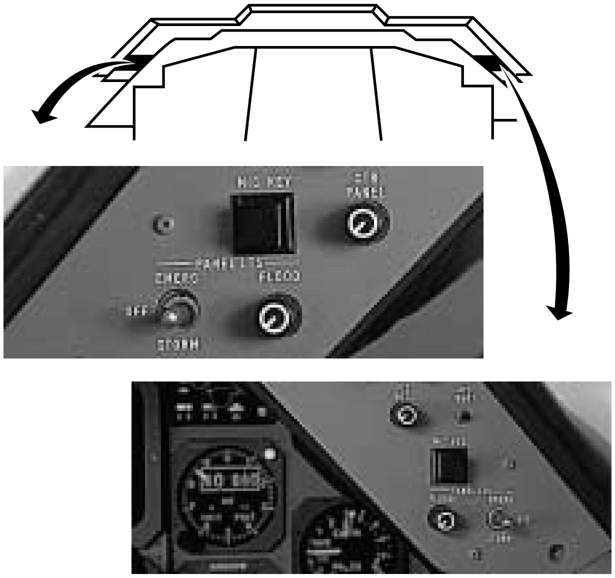


Figure 3-4. EMER–OFF–STORM Switch

PASSENGER COMPARTMENT LIGHTING

Passenger Safety Notices

FASTEN BELTS/NO SMOKING notices are installed in the cabin forward and rear bulkheads. A SEAT BELTS ON–OFF switch and a NO SMKG ON–OFF–AUTO switch are located on the cockpit roof panel (Figure 3-5). Turning either switch on illuminates the appropriate safety notice and activates an audio system chime unit. The chime also sounds as the notice is extinguishing. With the NO SMKG switch positioned to AUTO, the NO SMOKING notice automatically illuminates, and the chime operates as the nose landing gear locks down during extension.

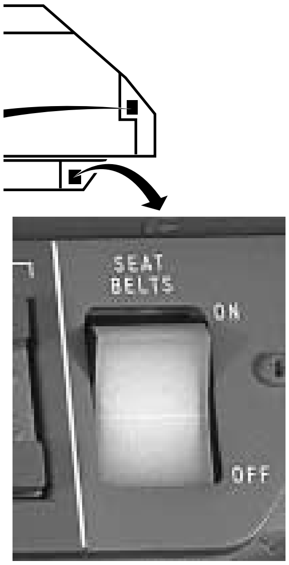


Figure 3-5. Passenger Warning Sign Switches

Vestibule Lighting

Entrance and vestibule illumination is provided by two step lights and a roof-mounted twin-light assembly. One step light mounted on the left side of the entrance and one light of the twin-light assembly are supplied directly from the No. 1 battery. The control switches (Figure 3-6) are wired for two-way switching. They are labeled "ENTRY DOOR" and "ENTRY LIGHTS" and are located on the upper inboard edge of the bulkhead panel and on the lower section of the left side forward cabinet rear face. The ENTRY DOOR switches also control the integral lighting for the refueling panel located at the rear of the bulkhead panel.

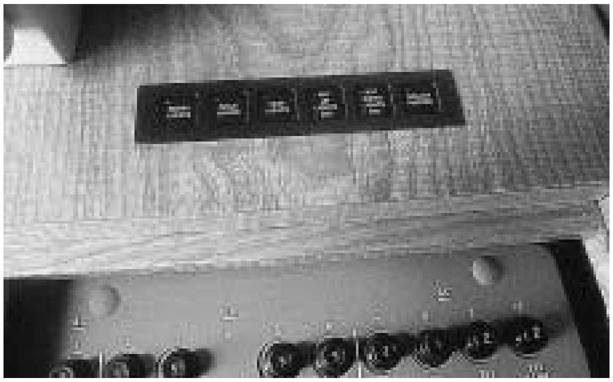


Figure 3-6. Vestibule Lighting Switch

EMERGENCY LIGHTING

Cockpit

Electrical power for emergency lighting is supplied from bus bar PE and is controlled by two EMERG–OFF–STORM switches on the pilot's and copilot's glareshields (Figure 3-7). Selection of the EMERG position on either switch illuminates lights located on the canopy rail and glareshield panel to provide main instrument panel lighting. A supply is also fed from bus bar PE to illuminate the standby compass integral light, which is normally controlled by the CTR PANEL dimmer switch on the left glareshield.

A box containing a number of spare bulbs is fitted at the base of the main distribution sidewall.

Passenger Compartment

Emergency Exit Lighting

An emergency light switch (Figure 3-8) on the interior lights section of the roof panel controls the DC power supply for lighting the main exit (entry door) and the red EXIT signs (Figure 3-9). The switch has positions labeled "ARM," "MANUAL," and "OFF."

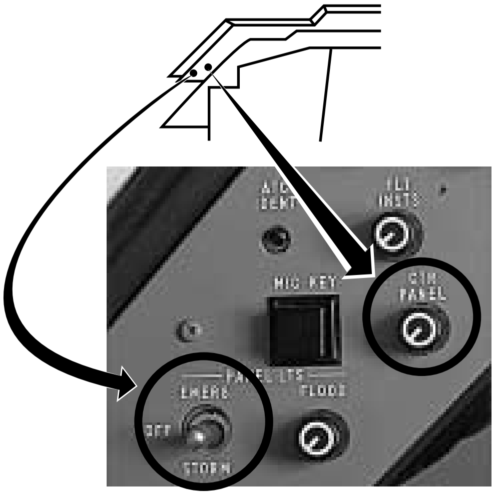


Figure 3-7. Emergency Lighting Controls—Cockpit

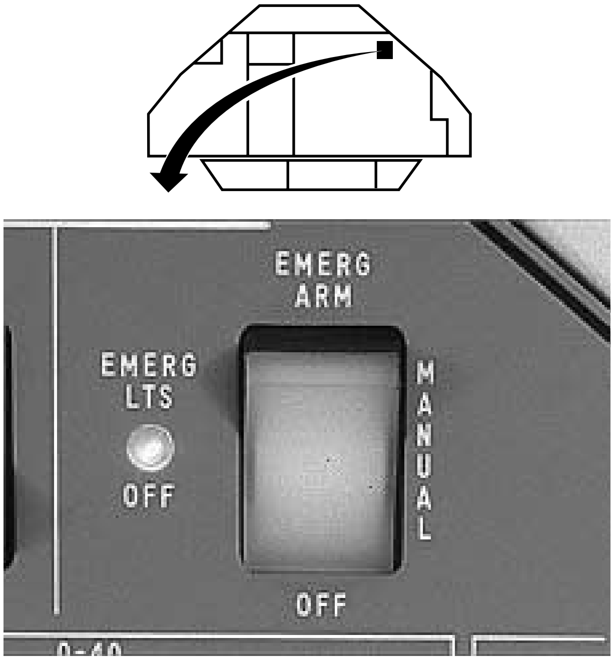


Figure 3-8. Emergency Light Switch

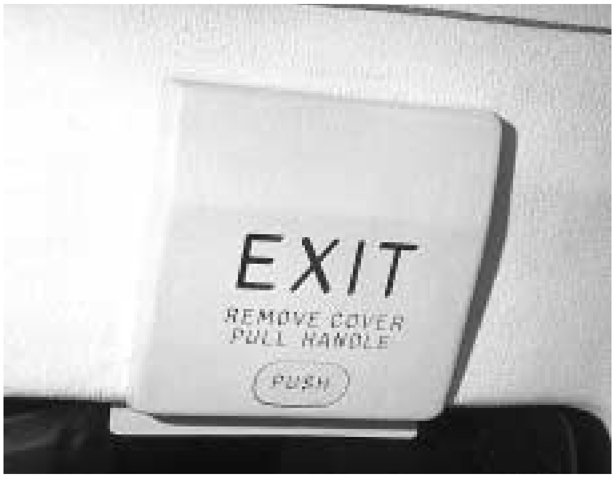


Figure 3-9. EXIT Light

With the aircraft BATTery switch ON and the MANUAL–OFF–ARM switch set to OFF, the EMRG LTS OFF annunciator is illuminated. Setting the switch to MANUAL illuminates the EXIT notices (Figure 3-10). With the MANUAL–OFF–ARM switch reselected to OFF, the EMRG LTS OFF annunciator is illuminated.

Selecting the switch to ARM (normal flight setting) extinguishes the EMRG LTS OFF annunciator and arms the system for automatic operation. During automatic operation, the EXIT notices illuminate in the event of PE busbar failure.

Alternate Emergency Exit Lighting

In accordance with FAR Part 125, these emergency lights are operated off two batteries, designated FWD and AFT.

- FWD:
  - Main door illuminated EXIT sign.
  - Six evenly divided aisle lights
- AFT:
  - Two lights on each side of emergency escape hatch
  - Escape hatch external lamp (overwing escape route)
  - External right wing leading edge to ground area lamp

These lights automatically illuminate when electrical power is removed from the PE busbar.

REAR EQUIPMENT BAY LIGHTING

A light and socket are provided in the roof of the rear equipment bay for general area illumination and power for a portable plug-in light (Figure 3-11).

Electrical DC power for light and socket is taken from battery No. 1 via an ON–OFF switch on the structure adjacent to the bay door hinge. Control of the switch is by manual selection to ON or OFF and automatic selection to OFF when the door is closed.

The light body incorporates a bulb, power socket, and a bar on–off switch. The bar switch

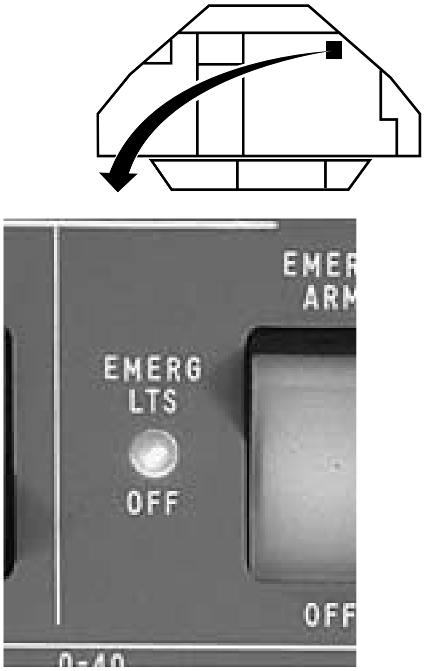


Figure 3-10. EMERG LTS OFF Light

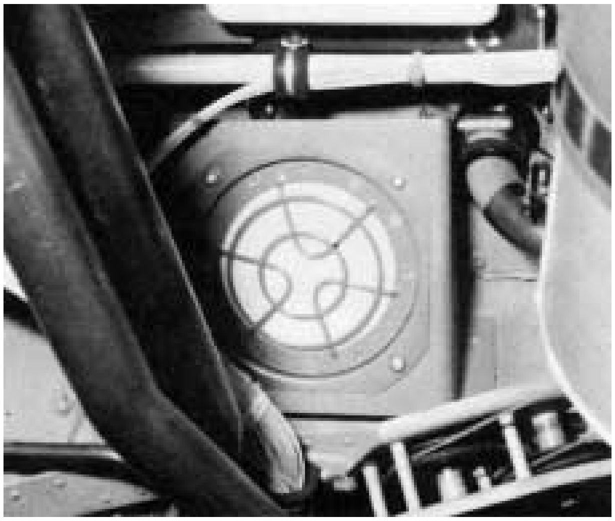


Figure 3-11. Rear Compartment Light

is wired in series with the main on–off switch and controls power to both the light and socket.

EXTERIOR LIGHTING

Exterior lighting consists of navigation, landing, taxi, anticollision beacons, strobe lights, and wing inspection spotlights. Exterior lighting locations are illustrated in (Figure 3-12). All exterior lighting control switches are located on the cockpit roof panel (Figure 3-13).

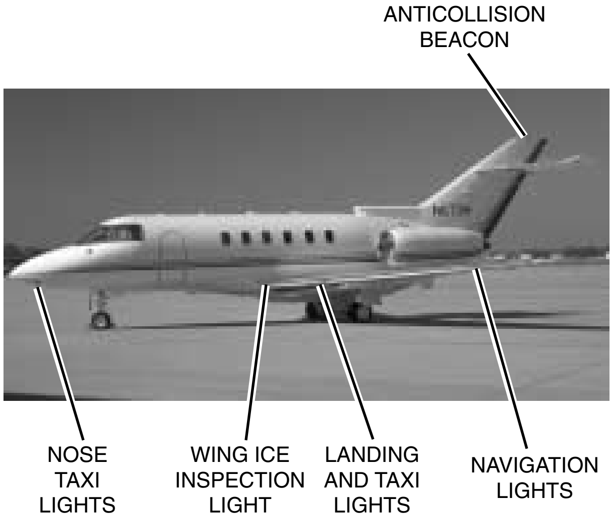


Figure 3-12. Exterior Lighting Locations

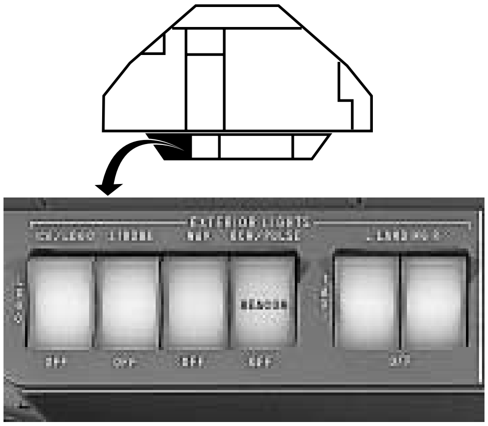


Figure 3-13. Exterior Lighting Controls

NAVIGATION LIGHTS

A navigation light is incorporated into each leading-edge wingtip, and one is installed in the aft tip of the fuselage. The light control switch is labeled “NAV” and “OFF” and receives power from the PE bus bar.

LANDING AND TAXI LIGHTS

Individual landing lights and taxi lights are installed in the leading edge of the wing. The lights are controlled by two switches labeled “L” or “R LANDING,” “TAXI,” and “OFF.” In TAXI position, the taxi lights are illuminated; in LANDING position, the landing lights are illuminated. The left wing landing light and taxi lights are powered by the PE bus bar; the right set, by the PS2 bus bar. The radome-mounted nose taxi lights are powered by PS1 left and PS2 right and will illuminate any time when either the wing landing lights or wing taxi lights are turned ON.

ANTICOLLISION BEACONS

A red anticollision beacon is located on the tip of the vertical stabilizer and another on the bottom center fuselage. Both lights are controlled by a switch labeled “BEACON” and “OFF.” The lights are powered by the PS2 bus bar. An option is available for a beacon/pulse system.

STROBE LIGHTS

Along with the standard airplane position light presentations, strobe lighting provides an additional means of airplane identification. Strobe lighting consists of three flashing white condenser discharge lights, a power supply unit for each light, and a control STROBE light switch on the flight compartment roof panel.

One strobe light is fitted in each wingtip and on the rear fuselage extremity.

WING ICE INSPECTION SPOTLIGHT

A light is installed on the left and right side of the fuselage and directed to illuminate the wing leading edge. The light is controlled by a switch labeled "ICE" and "OFF" and is powered by the PS1 and PS2 bus bars. An option is available for a logo light system.

ELECTRICAL POWER SOURCES

PE bus bar

- CB panels (spot)
- CB panels (wander)
- Chart (left and right)
- Instrument panels emer.
- Standby compass
- Navigation
- Left landing
- Left taxi

PS1 bus bar

- Wing inspection (left)
- Strobe
- Nose taxi (left)

PS2 bus bar

- Right landing
- Right taxi
- Beacons
- Nose taxi (right)
- Logo light (right)
- Logo light (left)
- Wing Inspection (right)

QUESTIONS

1. Power for the EMERG–OFF–STORM switch (EMERG position) is controlled by the:

A. PS2
B. PE
C. PS1
D. XE

2. The entry light switch is located on the:

A. Roof panel
B. Left bulkhead sidewall
C. Center console
D. Right bulkhead sidewall

3. If the NO SMKG switch is positioned to AUTO, the NO SMOKING light will illuminate when:

A. A downward force of 1.5 g is felt.
B. The nose gear locks down during extension.
C. Cabin altitude reaches 1,500 feet.
D. The vestibule light is turned on.

4. For normal operations, the EMERG lights switch should be positioned to:

A. MANUAL
B. OFF
C. ARM
D. EMERG

CHAPTER 4
MASTER WARNING SYSTEMS

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CHAPTER 4
MASTER WARNING SYSTEMS

INTRODUCTION

The master warning system consists of two master warning lights on the glareshield panel, a center annunciator panel, five roof annunciator panels, and a system of audible warnings. The capability to dim and test all annunciators is provided. The system provides a visual and/or audible indication of system status and directs the pilot's attention to any problem area.

GENERAL

The master warning system (MWS) is designed to include all serious warning indications on one panel. This panel is located on the center instrument panel as shown in Appendix B. Warnings that require immediate crew action illuminate red (warning) and cause two red (MWS) lights on the glareshield to flash. Pressing the face of either glareshield light will extinguish both lights and cause the warning annunciator light to dim. If the fault causing the annunciator to illuminate is corrected, the annunciator automatically extinguishes. Should the fault not be corrected, the annunciator remains illuminated but at the dimmed level. Any subsequent problem that illuminates an annunciator causes all annunciators to come to full brilliance.

The red glareshield lights will not operate when an amber annunciator illuminates. However, pressing either one of the MWS lights initiates the dimming circuit and reduces the intensity of any illuminated amber annunciator. As with red warnings, any subsequent caution light causes the annunciators to come to full brilliance if the fault is still in existence.

Indications of less urgency are illuminated in amber (caution) and are engraved to identify system association. The lights on the center annunciator panel labeled "FUEL XFD TFR," "ICE PROT," "ELECT," "FUEL," "DUCT OVHT," "ENG 1 FIRE," and "ENG 2 FIRE" are repeaters. Each is engraved with a vertical pointing arrow and, when illuminated, directs the pilot's attention to the appropriate section of the roof panel (Appendix B).

The DUCT OVHT light comes on when there is an overheat condition in the air-conditioning ducting. If the light comes on, it flashes and directs the pilot's attention to the air-conditioning system controls in the environmental section of the roof panel. When corrective action is not taken, the light will cycle on and off as the system cycles.

White lights are provided as an indication of completed operations and as system status indicators. They can be dimmed by pressing the face of either MWS light (these will not be flashing), but will come to full brilliance when any annunciator initially illuminates.

DC power to the master warning system and test circuits are supplied from bus bar PE.

MASTER WARNING LIGHTS

Two red lights, one in front of each pilot, are labeled "MWS" (Appendix B). They flash whenever a red warning annunciator is displayed or when the TEST switch is depressed. An uneven rate of flash indicates failure of one of the duplicated red warning circuits.

Pressing the face of either light cancels both red flashing lights. In normal conditions (even when the lights are not illuminated), this action also reduces lighting intensity of any red (except engine fire), amber, or white annunciator to a preselected level.

CENTER ANNUNCIATOR PANEL

The center annunciator panel is located on the center instrument panel and contains red warning, amber caution, and white advisory annunciators. The annunciators, their colors, and causes for illumination are given in Table 4-1.

Table 4-1. CENTER ANNUNCIATOR PANEL

Annunciator	Cause for Illumination
APU WARN	An annunciator on the APU panel is illuminated.
REVERSER	Ground—Rudder bias is not inhibited Flight—Autostow is activated.
MACH TRIM FAIL	Restrict speed to Mach 0.73 or less, unless autopilot is engaged.
HP AIR 1 OVHT HP AIR 2 OVHT	Indicated engine bleed-air temperature is excessive.
REAR BAY OVHT	An overheat condition exists in the rear equipment bay.
ENG 1 FIRE ↑ ENG 2 FIRE ↑	A fire or overheat condition is detected for the indicated engine.
CABIN ALTITUDE	Cabin altitude has reached 9,300 feet.
OIL 1 LO PRESS OIL 2 LO PRESS	Indicated engine oil pressure is less than 25 psi.
ELEV/AIL TRIM	The elevator/aileron is mistrimmed prior to takeoff.

Table 4-1. CENTER ANNUNCIATOR PANEL (Cont)

Annunciator	Cause for Illumination
HYD 1 LO PRESS HYD 2 LO PRESS	Pressure from the indicated hydraulic pump is 1,500 psi or less.
HYD OVHT	The main hydraulic system has overheated.
MAIN AIR VALVE 1 MAIN AIR VALVE 2	The indicated main bleed-air valve is not in the selected position (time delay of 50 seconds) or the valve is open while the airplane is on the ground.
AUX HYD LO LEVEL	The auxiliary hydraulic tank is not full.
ENG 1 CMPTER ENG 2 CMPTER	Steady light indicates the associated engine fuel computer has failed or the switch is in the OFF or OVSD PROT position
EMER BRK LO PRESS	The emergency brake accumulator pressure is 2,250 psi or less.
ENG 1 A/ICE ENG 2 A/ICE	The indicated engine's air is not available for anti-icing.
ICE PROT ↑	An annunciator on the ICE PROTECTION roof panel is illuminated.
ELECT ↑	An annunciator on either DC or AC power roof panels is illuminated.
FUEL ↑	A fuel roof panel annunciator is illuminated.
DUCT OVHT ↑	An overheat condition exists in the air-conditioning ducts.
ENT DOOR UNLOCKED	The entry door is not properly secured.
APU MSTR ON	The APU master switch is in the ON position.
RUDDER BIAS	One or both rudder bias switches are in the OFF position.
ICE PROT SELECTED	An engine ice protection system is selected.
FUEL XFD/TFR ↑	The WING FUEL XFD/TFR or AUX FUEL TFR light on the roof panel is illuminated.
STALL INDENT	An annunciator on the left or right stall identification panel is illuminated.

ROOF ANNUNCIATOR PANELS

There are four groupings of annunciators on the roof panel plus the engine fire lights. Under the appropriate heading, the annunciators, their colors, and causes for illumination are given in Tables 4-2, 4-3, 4-4, 4-5, and 4-6.

Table 4-2. ICE PROTECTION

Annunciator	Cause for Illumination
L SCREEN OVHT R SCREEN OVHT	Indicated windshield has reached the overheat limit.
ALTR 1 FAIL ALTR 2 FAIL	Indicated engine-driven alternator has failed.

Table 4-2. ICE PROTECTION (Cont)

Annunciator	Cause for Illumination
L VANE HTR FAIL R VANE HTR FAIL	Indicated vane heater has failed.
SIDE SCRN OVHT	Indicates left or right side window has overheated.
ICE DETECTED	Airframe ice has been detected.
PITOT HTR FAIL	Indicates loss of left or right pitot heat.
ANTICE LO PRESS	Indicates low-pressure output from airframe deicing pump.
ANTICE LO QTY	Indicates airframe deicing quantity low.
IGN ON	Indicates one or both continuous ignition circuits is energized.

Table 4-3. FUEL

Annunciator	Cause for Illumination
ENG 1 FUEL ENG 2 FUEL	Indicated fuel filter differential is excessive, or fuel temperature in the indicated engine fuel pumps is above normal.
REFUEL ON	Any valve in the refuel system is not in the closed position.
AUX FUEL TFR	Auxiliary fuel transfer lever is in the open position.
WING FUEL XFD/TFR	Wing fuel crossfeed transfer lever is in crossfeed or transfer position.
FUEL 1 LO PRESS FUEL 2 LO PRESS	Fuel pressure in the supply line to the respective engine is low.

Table 4-4. DC POWER

Annunciator	Cause for Illumination
BATT 1 OVHT BATT 2 OVHT	Either battery temperature is out-of-limits high. (Nicad batteries only.)
BATT 1 CNTCTR BATT 2 CNTCTR	Indicated battery is not charging or emergency contactor is open.
BATT 1 HOT BATT 2 HOT	Either battery temperature is high (Nicad batteries only.)
GEN 1 FAIL GEN 2 FAIL	Indicated generator has failed.
BUS TIE OPEN	The PS1 to PS2 bus-tie contactor is open.

Table 4-5. AC POWER

Annunciator	Cause for Illumination
XS 1 FAIL XS 2 FAIL XE FAIL	Indicated AC bus bar has failed.
INV 1 FAIL INV 2 FAIL	Indicated inverter has failed.
STBY INV ON	The standby inverter is selected on.

Table 4-6. ENGINE FIRE LIGHTS

Annunciator	Cause for Illumination
ENG 1 FIRE ENG 2 FIRE	A fire or overheat is detected for the indicated engine.

TEST FUNCTION

The TEST switch (Figure 4-1) is located adjacent to the center annunciator panel and is used to test electronic warning circuits and lights. Operation of this push-button switch illuminates all master warning system annunciators at maximum intensity and flashes the MWS red lights and annunciator panel repeaters.

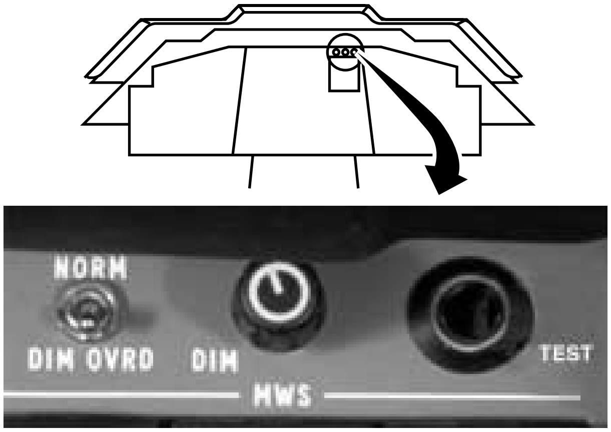

Figure 4-1. TEST Switch and DIM Controls

DIM FUNCTION

The dim function consists of a rheostat, a DIM OVRD-NORM switch, an MWS DIM FAIL ANN light, and a circuit tied to the red MWS lights. The controls are shown in Figure 4-1.

The rheostat is adjacent to the center annunciator panel and sets the annunciator's intensity level when in the NORM mode.

The DIM OVRD-NORM switch is for bypassing the dimming circuit in the event of a malfunction. Any malfunction is indicated by illumination of the MWS DIM FAIL ANN annunciator. With the DIM switch set to NORM, lighting intensity of all MWS annunciators (except ENG FIRE 1 and 2) can be varied by rotating the rheostat.

If an individual annunciator fails to operate correctly because of a dimmer circuit fault, the MWS DIM FAIL ANN annunciator illuminates. In this event, DIM OVRD must be selected. With OVRD set, the dimming circuit is bypassed, any illuminated annunciator goes to full bright intensity, and the MWS DIM FAIL ANN warning cancels. In this switch position, pressing the MWS red lights has no effect on the intensity of any annunciator.

SYSTEM OPERATION

When a system status change or fault occurs, the appropriate individual annunciator illuminates at maximum intensity. In the event of a red warning, both MWS red lights flash, and, if the red annunciator is located on the roof panel, the associated repeater illuminates steady. If the roof panel annunciator is amber, the repeater flashes.

Pressing either MWS red light reduces the annunciator's (and repeater's, if applicable) intensity to the lighting level selected by the MWS dimmer switch (rheostat), and the repeater, if flashing, changes to steady. Should an additional system status change or fault occur, the relevant annunciator illuminates at maximum intensity, and any dimmed annunciator becomes full bright. Subsequent dimming is achieved by pressing either MWS red light. This dim lighting level is maintained until the system fault clears and causes the annunciator to extinguish.

AUDIBLE WARNINGS

Bells, a horn, and an audio chime are generated to supply audible warnings for several airplane systems. The horn sound may be emitted either intermittently or continuously for different warnings of separate systems.

ENGINE FIRE DETECTION

Each engine fire overheat detector system is connected to an associated ENG FIRE warning light on the FIRE panel, an associated ENG FIRE repeater light on the MWS panel, the MWS flashers, an HP cock warning light, and, via a relay, a warning bell. Bell operation may be canceled by setting a switch to BELL CNCLD (Figure 4-2).

Each detector circuit is provided with a FIRE warning TEST push switch. Pressing the switch operates the associated warning system.

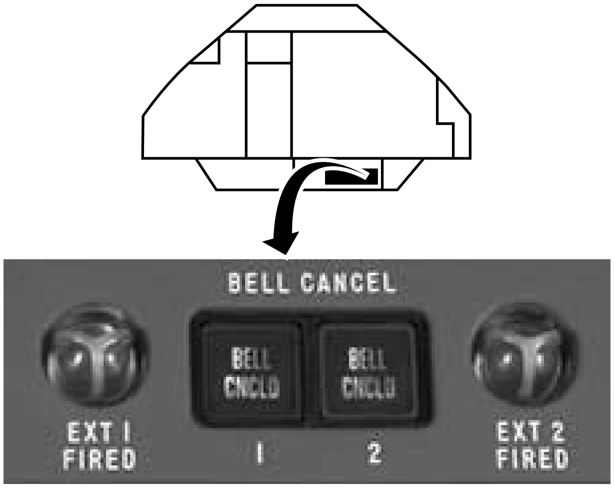

Figure 4-2. Fire Bell Isolate Switch

LANDING GEAR HORN WARNING

The horn gives a continuous steady note when the gear is in any position other than down and locked and the flap handle is in landing or approach position. On the ground, with the landing gear down and locked, the gear selector lever out of the down position, and start power selected, the horn also sounds. In either circumstance the horn cannot be overridden.

AIRBRAKES HORN WARNING

The horn gives a continuous steady note if the gear is down and locked, both throttles are above approximately 60-70% N_1 rpm, and the airbrakes are open.

THROTTLES BELOW 60%–70% N_1 RPM

When the landing gear is in any position other than down and locked when either throttle is moved below approximately 60-70% of N_1 rpm, and the airspeed is below 150 knots, the horn will emit a continuous steady note. Operation of the horn CANCEL button (Figure 4-3) on the No. 1 throttle lever interrupts power to the horn. The horn remains silent until the throttles are both above approximately 60-70% N_1 rpm or the airspeed is above 150 knots.

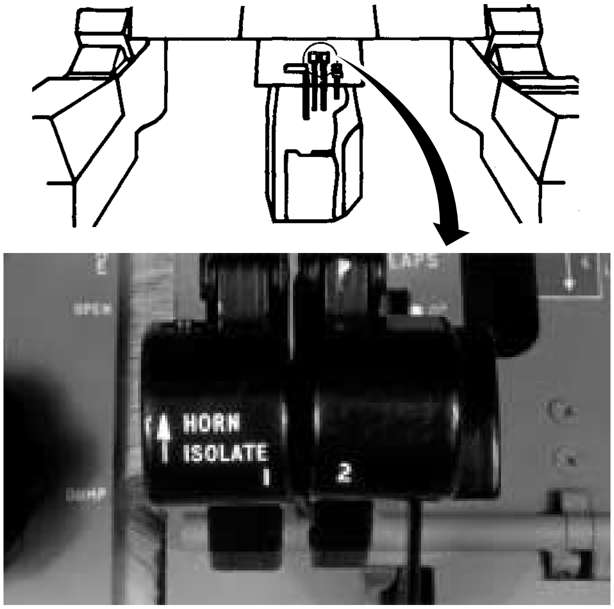

Figure 4-3. Landing Gear Horn Cancel Button

SPEED WARNING

An intermittent horn warning will occur if the airplane is flown above the prescribed speed limits, i.e., 280 knots (with fuel in the ventral tank) or 335 knots/0.8 M knots (with the ventral tank empty).

An air data computer system initiates this warning.

TESTING

Four push switches are located on the test section of the roof panel-one for the low V_{MO} and one for the V_{MO}/M_{MO} of each of the air data computers.

CABIN DEPRESSURIZATION

If the cabin altitude rises above 9,300 feet, a pressure-operated switch closes. This causes:

- The warning horn to sound (continuous steady note)
- The red CABIN ALTITUDE light on the master warning panel to illuminate at full brilliance
- The red MWS annunciators on the glareshield to flash

NOTE

With flaps lowered beyond 15°, the cabin altitude warning horn is electrically isolated.

To cancel the warning:

- Press the horn isolate button on the No. 1 throttle lever.
- Press either red MWS annunciator flashing light on the glareshield. This stops the flashing and causes the CABIN ALTITUDE light to dim to a preset level.

AIRFRAME DEICING

Anytime the airframe deicing timer switch returns to zero, an audio tone sounds, indicating pump shutoff.

ELECTRICAL POWER SOURCES

PE bus bar

- Master warning supply 1
- Master warning supply 2
- Master warning test

QUESTIONS

1. The MWS annunciators can be dimmed:
   A. With the DIM OVRD switch
   B. By turning the MWS dimmer clockwise
   C. By pressing the face of either red MWS glareshield annunciator
   D. By individual system dimmers

2. With the exception of DUCT OVHT, the arrows on the center annunciator panel's flashers signify:
   A. That the indicated system's temperature is increasing
   B. The direction to the glareshield flashers
   C. That the indicated system's temperature needs to be increased
   D. That a roof panel annunciator is illuminated

3. The white annunciators indicate:
   A. Completed operations and system status
   B. That the associated system is off
   C. A system malfunction
   D. A system failure

4. The bus bar that powers the master warning system is:
   A. PS1
   B. PE
   C. PS2
   D. XE

5. The master warning lights can be extinguished by:
   A. Resetting the MWS circuit breaker
   B. Pressing the TEST switch
   C. Pressing the face of either red MWS annunciator
   D. Use of the DIM OVRD switch

6. To illuminate all MWS annunciator lights:
   A. Press the MWS TEST switch.
   B. Position the DIM OVRD switch to OVRD.
   C. Press the face of either MWS annunciator.
   D. Turn the MWS dimmer full clockwise.

7. To bypass the DIM circuit:
   A. Position the DIM OVRD switch to NORM.
   B. Press the MWS DIM FAIL ANN light capsule.
   C. Position the MWS dimmer fully clockwise.
   D. Position the DIM OVRD switch to OVRD.

CHAPTER 5
FUEL SYSTEM

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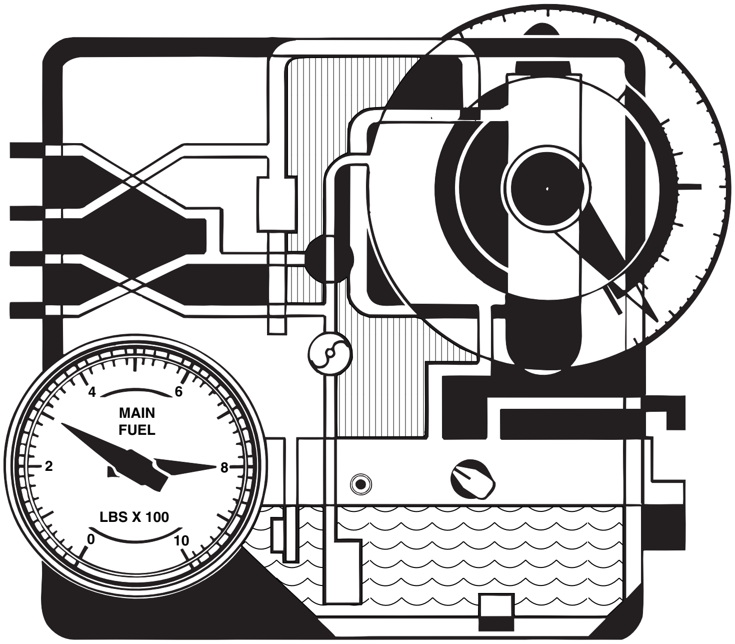
FUEL SYSTEM



INTRODUCTION

The Hawker 800 XP fuel system includes left and right integral wing tanks, and an auxiliary ventral tank located in the rear fuselage.

This chapter covers the operation of the fuel system up to the engine-driven fuel pump. At that point, fuel system operation becomes a function of the engine. Refer to Chapter 7, “Powerplant,” for additional information.

GENERAL

One electric boost pump in each wing tank delivers fuel under pressure to its respective engine during normal operating conditions. Each boost pump operates two jet pumps. The jet pumps transfer fuel from the outer portion of the wing tanks and the ventral tank to the fuel sump compartment where the boost pump is located. This also ensures an adequate fuel reservoir for the boost pump. Fuel may be transferred from the ventral tank into both wing tanks. The fuel system has both gravity- and pressure-refueling/defueling capabilities. All tanks can be pressure-refueled or defueled simultaneously.

FUEL TANKS

GENERAL

Fuel is stored in the left and right integral wing tanks and an auxiliary ventral tank located in the lower part of the aft fuselage.

WING TANKS

The wing tanks are formed by sealing the major portion of the wing structure (Figure 5-1). The tanks are divided by ribs and spars to provide antisurge compartments. Flapper valves prevent the fuel from moving outboard into the wing. The number one (fuel sump) compartment is the lowest area in each tank, allowing it to act as a reservoir. The number one compartment contains an electrical boost pump, two jet pumps, a water drain, and a fuel drain.

VENTRAL TANK

The auxiliary ventral tank is located in the lower part of the aft fuselage. It has a water drain and a fuel drain.

VENT SYSTEM

Vent pipes from the wing tanks and ventral tank allow venting to surge tanks in the wingtips. Each surge tank incorporates a NACA anti-icing flush-mounted ram-air vent assembly on the wing undersurface. The ram-air intake of the vent provides a slight positive pressure throughout the vent system during flight. A water drain is provided for each surge tank. The ventral tank vents into the left wing surge tank. (Refer to Figure 5-8, this chapter.)

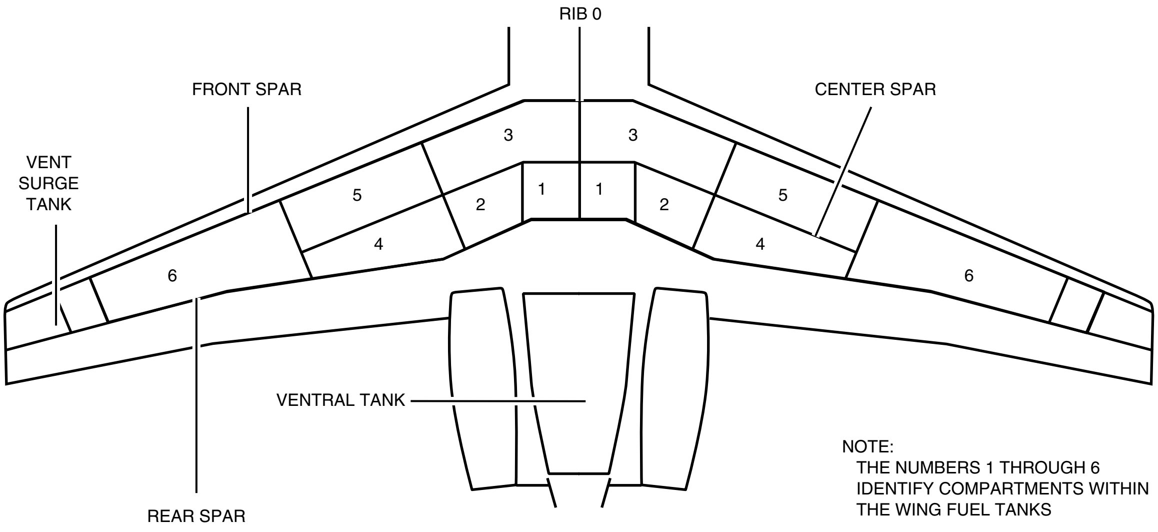
CAPACITIES

The wing tanks each have a usable capacity of 631 U.S. gallons. The ventral tank has a usable capacity of 224 U.S. gallons. The total usable fuel is 1,486 U.S. gallons or 10,000 pounds. Table 5-1 lists fuel capacities.

For flight, the ventral tank must be completely filled or empty, but intermediate fuel loads are acceptable for the wing tanks and can be obtained by manually selecting their refuel valves off at the required contents during refueling.

FUEL DISTRIBUTION

The fuel feed system to both engines is similar. Fuel is drawn into each number one compartment by two jet pumps activated by the respective boost pump (Figure 5-2). One jet pump draws fuel from the various wing tank compartments. The other jet pump draws fuel,



NOTE:
THE NUMBERS 1 THROUGH 6 IDENTIFY COMPARTMENTS WITHIN THE WING FUEL TANKS

Figure 5-1. Fuel Tanks

Table 5-1. FUEL CAPACITY

TANK	IMP. GAL	U.S. GAL	LITERS
Wing left	526	631	2,391
Wing right	526	631	2,391
Ventral	187	224	850
Totals	1,239	1,486	5,632

NOTE:

When gravity refueling is used the tank capacities will increase by 2 Imperial gallons (3 U.S. gallons or 7 liters).

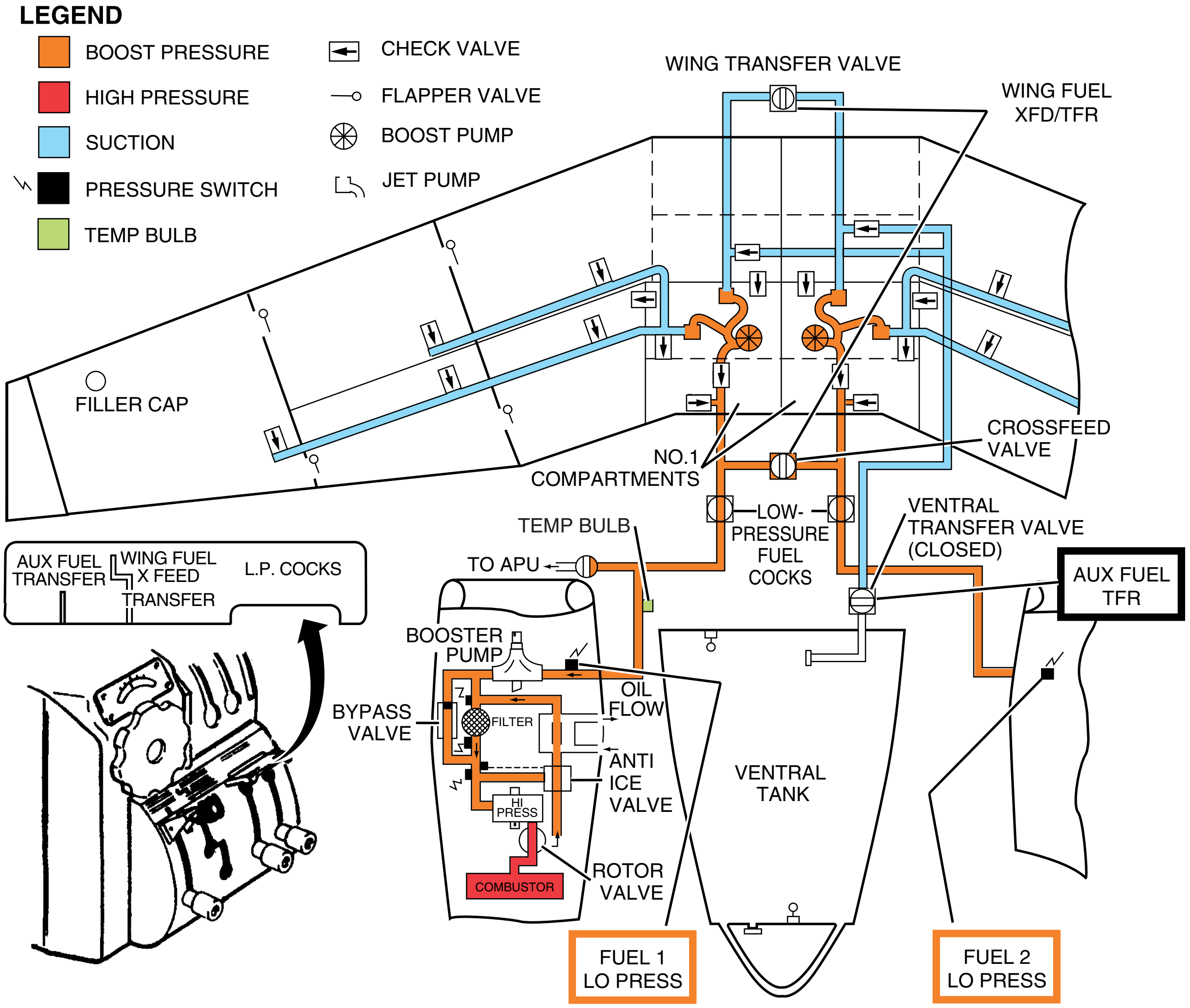


Figure 5-2. Fuel System

either from the opposite wing tank or from the ventral tank, depending on the selection of the appropriate transfer valves. It cannot draw any fuel if both transfer valves are closed.

Under normal conditions, each electric boost pump delivers fuel under pressure to the engine on the same side via a low-pressure fuel cock. Should both electric boost pumps fail, fuel gravitates into each number one compartment. This enables the engine-driven fuel pumps to draw fuel for operation by suction feed.

With both electric boost pumps operating and the ventral transfer valve open, fuel from the ventral tank transfers to both wings due to jet pump operation. Fuel transfer takes about 12 minutes. With a single electric boost pump operating, fuel transfers to the wing tank containing the operating pump; transfer time, in this condition, may increase to more than 20 minutes. Should both electric boost pumps become inoperative, ventral tank fuel cannot be transferred.

Fuel in the left wing tank is normally supplied to the number one engine and auxiliary power unit (APU), and fuel from the right wing is supplied to the number two engine. Fuel from both wing tanks can be made available to either or both engines through the crossfeed valve. In the event of single electric boost pump failure, opening the crossfeed valve enables the operating pump to supply both engines. The wing transfer valve allows fuel to be drawn from one wing tank to the other.

The low-pressure fuel cocks, one in each supply line, isolate the engine fuel feed pipes from the tanks.

Pressure switches, tapped into each engine fuel line, are connected to FUEL 1 and 2 LO PRESS annunciators (Appendix B). Illumination of these annunciators during normal fuel operation indicates a loss of pressure and signals a possible electric boost pump failure. A temperature bulb permits a fuel temperature indication to be read from the number one engine fuel supply line. The fuel in each engine fuel supply line passes through a flowmeter on each engine.

CONTROLS AND INDICATORS

CONTROLS

The electric boost pumps are controlled by three-position switches (ON, OFF, EMERG) on the FUEL section of the roof panel (Figure 5-3). The left and right electric boost pumps are normally supplied from bus bars PS1 and PS2, respectively, with the pump switch on. When the pump switch is on and the start power switch depressed (ON), the power source switches to PE. The pumps are connected to bus bar PE when EMERG is selected.

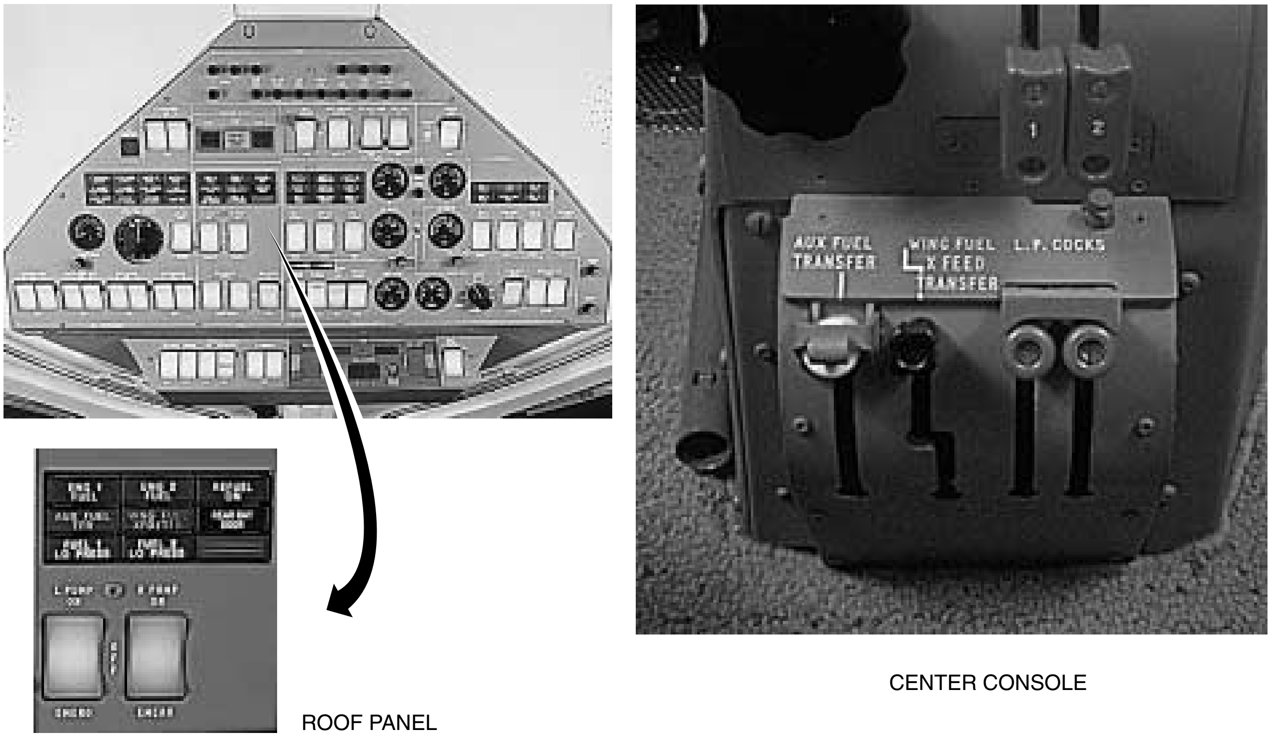
Two LP levers, an AUX FUEL TRANSFER lever, and a single WING FUEL X FEED TRANSFER lever are provided. The control levers are cable connected to low pressure, auxiliary, wing transfer, and crossfeed valves.

The control levers are located on the aft part of the center console. Each lever moves in a gated quadrant. Before selection can be made, the knob on each lever must be pulled out against spring pressure. The lever locks in the selected position when the knob is released. The WING FUEL X FEED TRANSFER lever is X shaped for ease of identification.

A ventral tank transfer valve is operated by the single AUX FUEL TRANSFER lever. A hinged guard fits over the lever in the closed (up) position. This guard must be lifted before the lever can be moved to the open (down) position.

The WING FUEL X FEED TRANSFER lever has three positions: closed, X FEED, and TRANSFER. Fuel from the left wing is supplied to the No. 1 engine, and fuel from the right wing is supplied to the No. 2 engine when the lever is in the closed position.

Moving the lever to the X FEED position opens the crossfeed valve. Fuel from both wings is available to either or both engines. A nonreturn valve in the outlet of each pump prevents fuel being transferred from one wing tank to the other during crossfeeding.



ROOF PANEL

CENTER CONSOLE

Figure 5-3. Fuel Controls

Moving the lever to the TRANSFER (bottom) position opens the transfer valve, and the crossfeed valve remains open. Fuel transfer from one wing to the other is achieved, in normal conditions, by switching off one electric boost pump. Fuel is drawn into the tank containing the operating pump by jet pump operation.

Two LP COCKS control levers are on the right side of the center console. The two levers control two low-pressure fuel valves, one in each engine supply line, which can isolate the engine fuel feed lines from the fuel tanks. Each valve is open when the associated LP COCK control lever is in the up position. A hinged guard, fitted above the control levers, must be lifted before the levers can be moved to the closed (down) position.

INDICATORS

Two amber annunciators, labeled “FUEL 1 LO PRESS” and “FUEL 2 LO PRESS,” are located on the FUEL section of the roof panel (see Appendix B). The annunciators illuminate when the fuel pressure in the engine fuel line falls below 7 psi. The illumination of either light is repeated by an amber annunciator on the MWS center panel labeled “FUEL↑.”

A white AUX FUEL TFR annunciator located on the FUEL section of the roof panel illuminates whenever the AUX FUEL TRANSFER lever is moved to the open position. This advisory is repeated by a white annunciator on the MWS center panel labeled “FUEL XFD TFR↑.” (See Appendix B.)

An amber REFUEL ON annunciator located on the roof panel illuminates whenever any refuel valve is open. This advisory is repeated by an amber annunciator on the MWS center panel labeled “FUEL↑.” (See Appendix B.)

A white WING FUEL XFD/TFR annunciator located on the roof panel illuminates when the WING FUEL X FEED TRANSFER lever is moved out of the closed position. This advisory is repeated by a white annunciator on the MWS center panel labeled “FUEL XFD/TFR ↑.” (See Appendix B.) In the TRANSFER position the light flashes.

Two fuel quantity indicators, one for each wing, are located on the center instrument panel (Figure 5-4). They indicate fuel quantity in pounds times one thousand. Seven transmitter units in each wing are electrically connected to each fuel quantity indicator.

A magnetic fuel quantity indicator is provided for the ventral tank. The indicator is located between the two fuel quantity indicators and shows FULL, crosshatch, and EMPT. The high- and low-level float switches send signals to the indicators when the tank is full and empty, respectively. The indicator shows crosshatch anytime the tank is anywhere between full and empty and when electrical power is off. A small residual amount of usable fuel will remain in the tank after the ventral tank quantity indicator shows EMPT during the climb. The AUX FUEL TRANSFER lever should be left in the open position for 5 minutes after transition from climb to cruise to ensure that fuel transfer is complete. The ventral tank low-level float switch maintains a circuit to provide low VMO speed warning at 280 KIAS until ventral tank fuel has been transferred.

REFUELING

The airplane can be gravity refueled through a filler cap in the outboard section of each wing and in the left rear side of the ventral tank (Figure 5-5). A rubber flapper valve is fitted beneath each wing refuel cap to close the outlet in the event of an improperly closed cap.

Pressure refueling and defueling are provided by a single-point connection on the right side of the airplane below the right engine. Pressure refueling is controlled from a refueling control panel (Figure 5-6). The panel is located adjacent to the vestibule right-hand bulkhead; the panel illumination is controlled by the door ENTRY LIGHT and vestibule ENTRY LIGHT switches. Electrical power for fuel valve positioning is available directly from the PE bus bar or battery No. 1 through the refueling control panel ON-OFF power switch.

An option is also available for an aft external refueling panel.

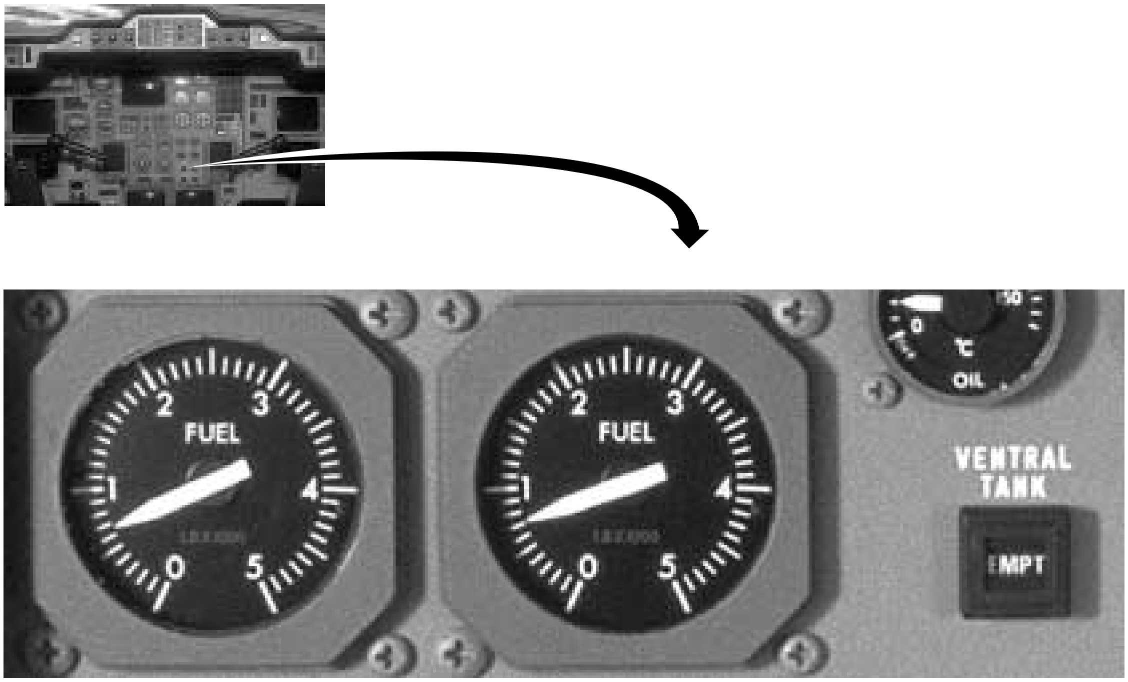


Figure 5-4. Fuel Indicators

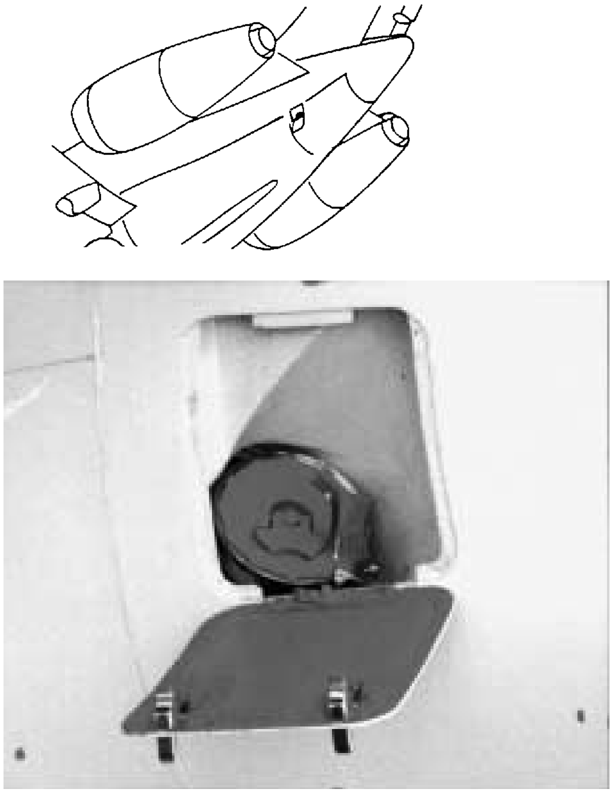


Figure 5-5. Ventral Tank Filler Cap

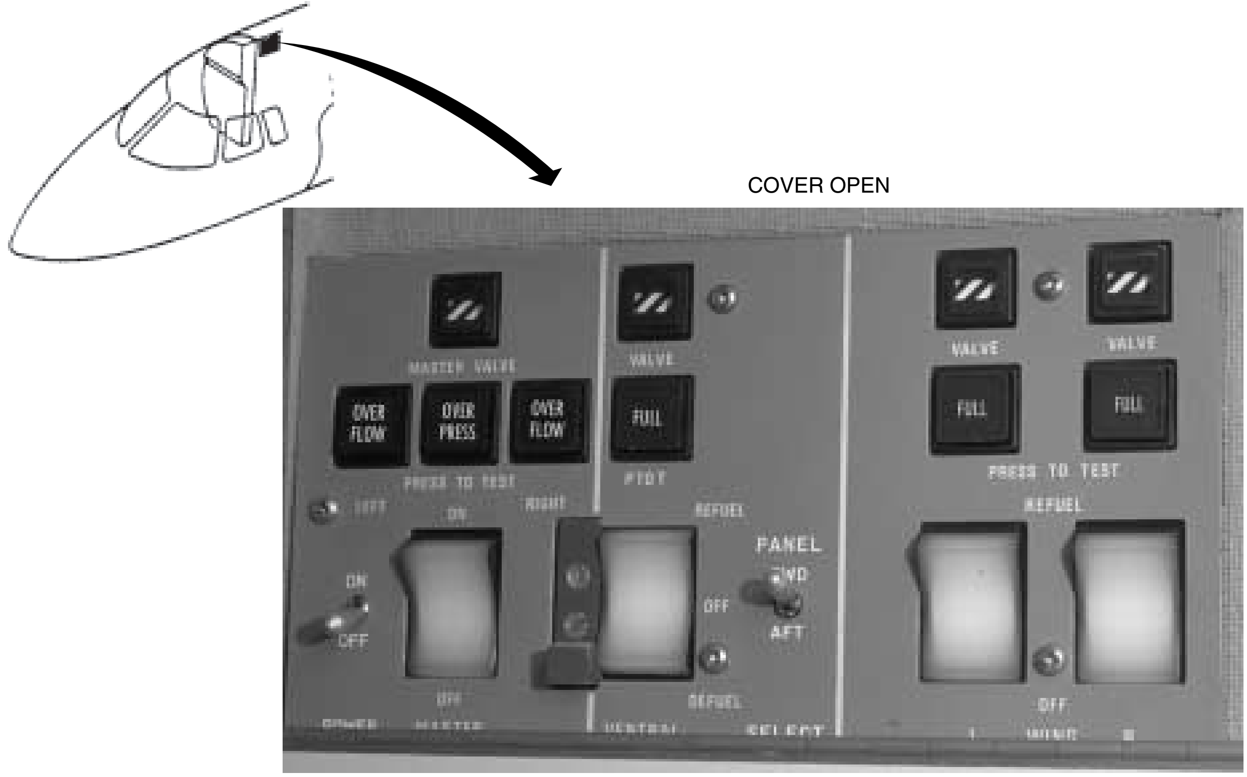


Figure 5-6. Refueling Control Panel

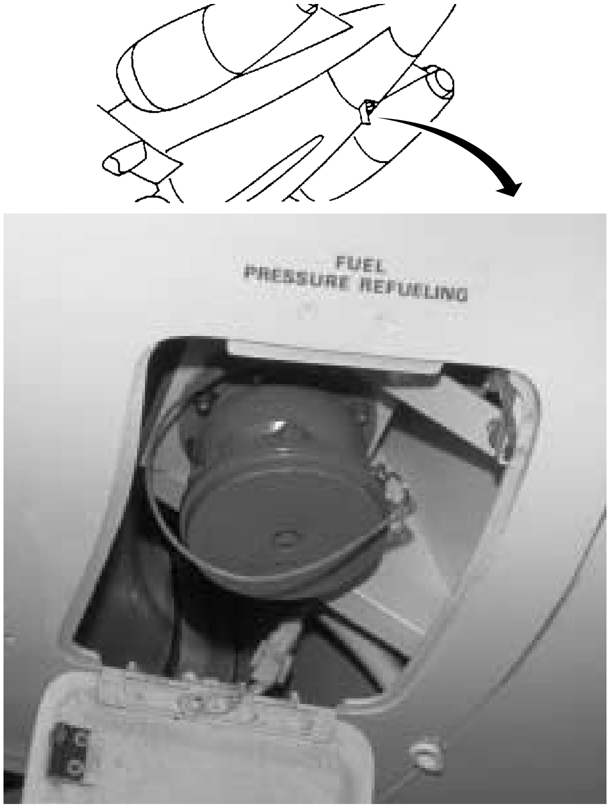


Figure 5-7. Pressure Refuel/Defuel Coupling

Fuel is loaded or off-loaded through a self-sealing coupling, accessible through a hinged door recessed into the ventral tank on the right side of the airplane (Figure 5-7). A pressure-reducing valve inside the coupling reduces refueling pressure within the system. Restrictors (within the low-pressure cocks) and surge relief valves eliminate transient high pressure.

NOTE

Part of the fuel system pipelines are common to both the fuel feed and refuel/defuel systems. Because of the location of the surge relief valves, the low-pressure cocks must be open when refueling any tanks or defueling the wing tanks.

Thermal-relief valves are installed in the refueling lines to provide protection against thermal expansion of the fuel (Figure 5-8).

A pressure switch in the line to the master refuel valve closes the master refuel valve and any open tank refuel valves if an overpressure situation is detected. A valve-shut indication is given at the refuel control panel.

Fuel flow into or from the system is controlled by an electrically operated master refuel valve. Additionally, each wing tank and the ventral tank has an electrically operated refuel valve. Each tank refuel valve can be mechanically positioned in the event of electrical failure.

Magnetic indicators on the refueling control panel show SHUT, crosshatch, or OPEN for each of the valves in the system. As each tank is filled, its high-level float switch causes the related tank valve to close. The tank valve indicator on the refuel control panel will show SHUT, and the tank FULL indicator will illuminate. The valve indicators show crosshatch when electrical power is off.

Should a malfunction cause any tank to overflow into its surge tank, a float-operated switch in the surge tank causes the master refuel valve and any open tank refuel valves to close. The appropriate OVERFLOW indicator on the refuel control panel will illuminate.

All tanks may be pressure refueled simultaneously. Intermediate fuel loads are acceptable for the wing tanks and can be obtained by selecting the refuel switch off at the required quantity level.

Refer to the appropriate airplane manual for refueling and defueling procedures.

ELECTRICAL POWER SOURCES

PE bus bar

- Left fuel booster pump-emer-during engine start
- Right fuel booster pump-emer-during engine start

PS1 bus bar

- Left fuel booster pump-on
- Left fuel flow/fuel used indicator

PS2 bus bar

- Right fuel booster pump-on
- Right fuel flow/fuel used indicator

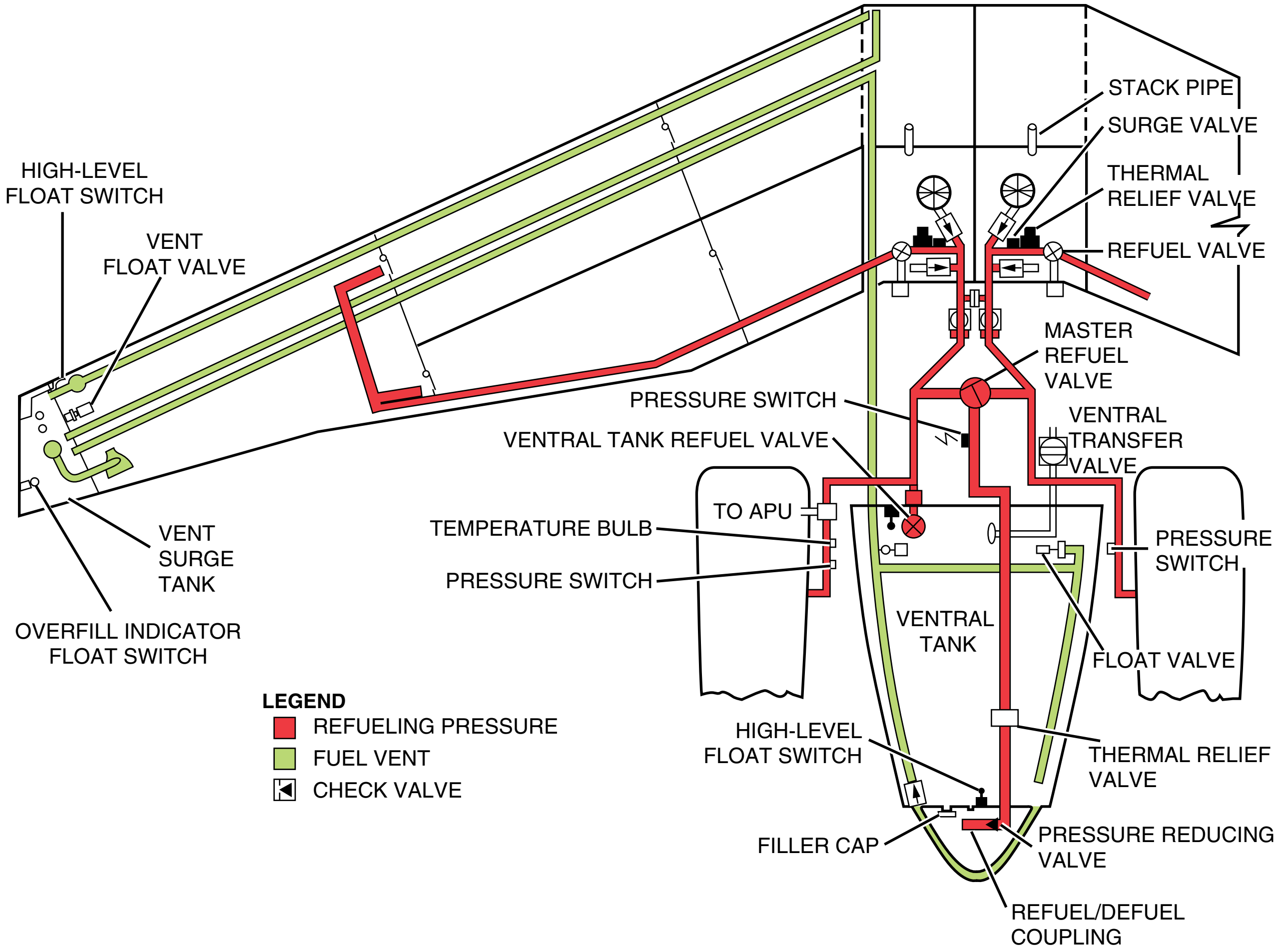


Figure 5-8. Refueling System and Vent Lines

LIMITATIONS

FUEL LOADING

Fuel loaded into the wing tanks shall be equally distributed between the two wing tanks.

Fuel must not be carried in the ventral tank unless each wing tank contains at least 3,450 pounds of fuel.

Before flight, the ventral tank must be completely filled or empty.

FUEL MANAGEMENT

During flight, including takeoff and landing, the maximum difference in fuel quantity between the wing tanks is 500 pounds.

Fuel carried in the ventral tank shall be transferred into the wing tanks when the fuel level in the wing tanks falls to 3,300 pounds.

The airplane must not be landed with fuel in the ventral tank except in an emergency.

QUESTIONS

1. All fuel annunciators and contents indicators are supplied from bus bar(s):

A. PE and/or BATT 1
B. PS1 and PS2
C. PE and XE
D. A and B

2. The number of water drains in the fuel system is:

A. Two
B. Three
C. Four
D. Five

3. When the ventral tank contains fuel, a low VMO horn will sound whenever airplane speed exceeds:

A. 270 KIAS
B. 280 KIAS
C. 292 KIAS
D. 320 KIAS

4. Total usable fuel capacity is:

A. 9,300 pounds
B. 9,440 pounds
C. 9,640 pounds
D. 10,000 pounds

5. The maximum lateral imbalance permitted is:

A. 250 pounds
B. 500 pounds
C. 750 pounds
D. 1,000 pounds

6. With both boost pumps inoperative:

A. Leave the AUX FUEL TRANSFER lever open.
B. Fuel will gravity transfer.
C. Both A and B
D. Ventral tank fuel cannot be transferred.

7. If the ventral tank is to be used in flight, it must be full, and each wing must contain more than:

A. 3,300 pounds of fuel
B. 1,000 pounds of fuel
C. 3,450 pounds of fuel
D. 2,000 pounds of fuel

8. The aircraft fuel system accepts a maximum refueling supply pressure of:

A. 25 psi
B. 45 psi
C. 50 psi
D. 55 psi

CHAPTER 6
AUXILIARY POWER UNIT

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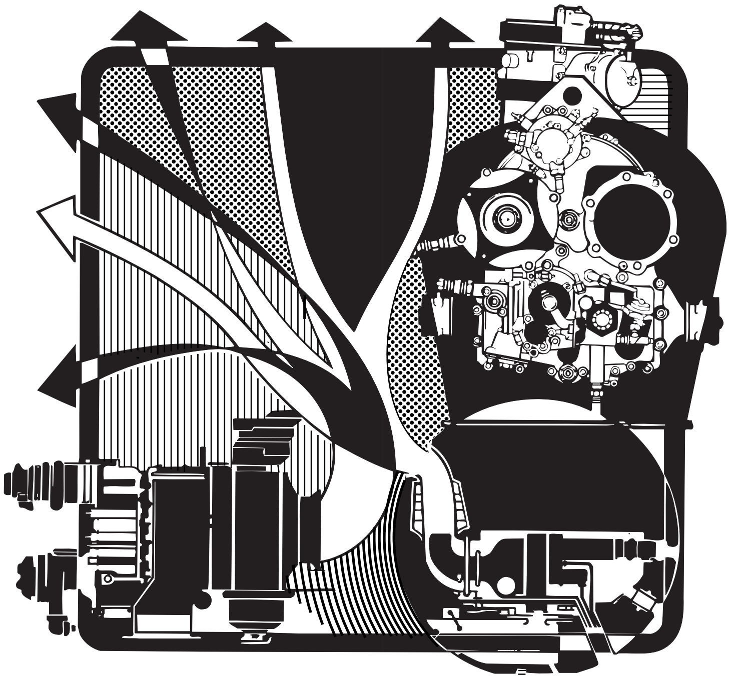
CHAPTER 6
AUXILIARY POWER UNIT



INTRODUCTION

The APU is a self sufficient unit installed in the rear equipment bay. Its function is to drive a DC generator to power the electrical system, charge the aircraft batteries, and provide bleed air to the aircraft environmental system for heating or cooling. Separate fire detecting, warning, and extinguishing systems are provided for the APU.

GENERAL

The Hawker 800 XP airplane may have an APU manufactured by the Garrett Turbine Engine Company GTCP 36-150, or by Solar Turbines Incorporated T62T-40. Some installations are approved for in-flight use without restrictions. Both APUs can be used in flight in an emergency. Since there are operating variations, a particular APU may not be fully covered within this chapter. Please consult your APU supplement to the *AFM* for specific operating instructions and restrictions.

GARRETT APU (GTPU 36-150W)

NORMAL PROCEDURES

This APU is equipped with a starter/generator and is approved for operation on the ground and in flight.

Operating Limits

Maximum RPM 110%

Maximum Exhaust
Gas Temperature (EGT) ....732°C
(operating)

Minimum oil temperature for starting APU:
Type II oil per MIL-L-26699 ....–40°C
Type I oil per MIL-L-7808 ....–55° C

Altitude:
Operating..........................0 – 30,000 ft
Starting ....0 – 20,000 ft

Start Cycles

The following start cycles are permitted when using aircraft batteries or ground power:

Starting:

1. Maximum EGT
   (transient) ....760°C to 870°C
   (15 seconds maximum duration)

   Maximum EGT
   (transient) ....870°C to 974° C
   (10 seconds maximum duration)

2. Hung Starts:
   (below 95% RPM)................10 seconds
   maximum duration

Maximum Permissible Electrical Load:

The maximum permissible electrical load on the Lear Siegler 23080-015 starter/generator is 300 amp (maximum).

Shut Down

Shut down APU if READY TO LOAD light does not illuminate within 60 seconds of start.

Do not operate APU if fire extinguisher agent container is depleted below:

- 300 psig at –40°F
- 575 psig at +70°F

Do not attempt to start APU if both main engine generators have failed.

Do not operate APU bleed air and main engine bleed air simultaneously. Both systems may be operated brief ly (1 minute maximum) during change over from one system to the other.

The APU is not approved for use in icing conditions.

Required Placards (Adjacent to APU Panel):

APU LIMITATIONS
OPERATING ....0 – 30,000 ft
STARTING ....0 – 20,000 ft

MAJOR SECTIONS

The APU consists of the following six main sections.

1. Air inlet
2. Compressor
3. Combustion
4. Turbine
5. Exhaust
6. Accessory Drive

CONTROL AND INDICATIONS

Normal Procedures

The APU provides a source of air for air conditioning and 28.0 volt DC power for energizing the airplane's DC electrical system. A description of the electrical system follows.

The APU utilizes fuel from the LH. fuel tanks. The LH aircraft fuel boost pump must be turned ON prior to the APU starting cycle.

The APU controls and annunciator lights are located an a single panel mounted behind the pilot seat.

APU generator is on line whenever APU is running (see Figure 6-1 or 6-2).

APU Indications

The APU control panel (Figure 6-2) is located on the flight compartment overhead panel and contains essential operator switches and indicators. Panel switches and indicators are as follows:

- ***DC LOAD Indicator***—Indicates current being generated by the APU.
- ***RPM Indicator***—APU revolutions per minute.
- ***EGT Indicator***—APU exhaust gas temperature (EGT °F).
- ***MAIN TEST OVV Switch***—Tests generator control units ability to trip APU generator off-line.
- ***LAMP TEST Switch***—Insures all panel warning lamps are operable.
- ***GEN Switch***—Manually controls generator function.
- ***BLEED AIR Switch***—Opens and closes load control switch.
- ***MASTER Switch***—Provides power to APU system and initiates start sequence.

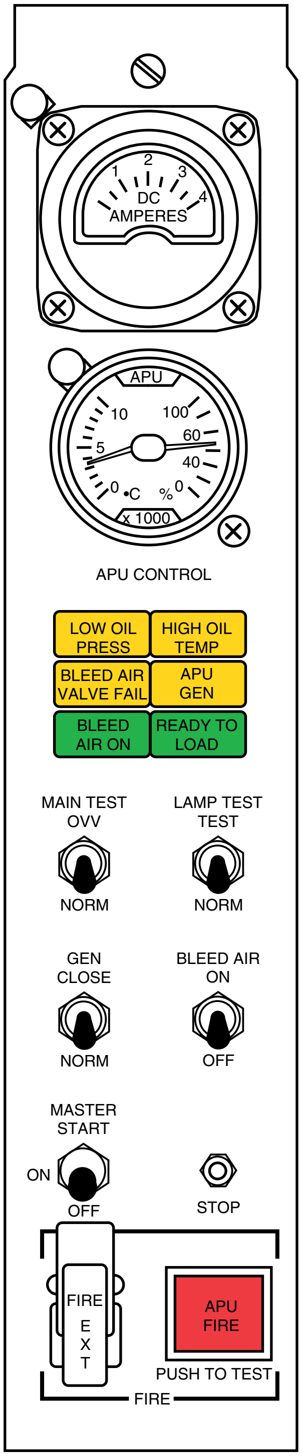


Figure 6-1. Garrett APU

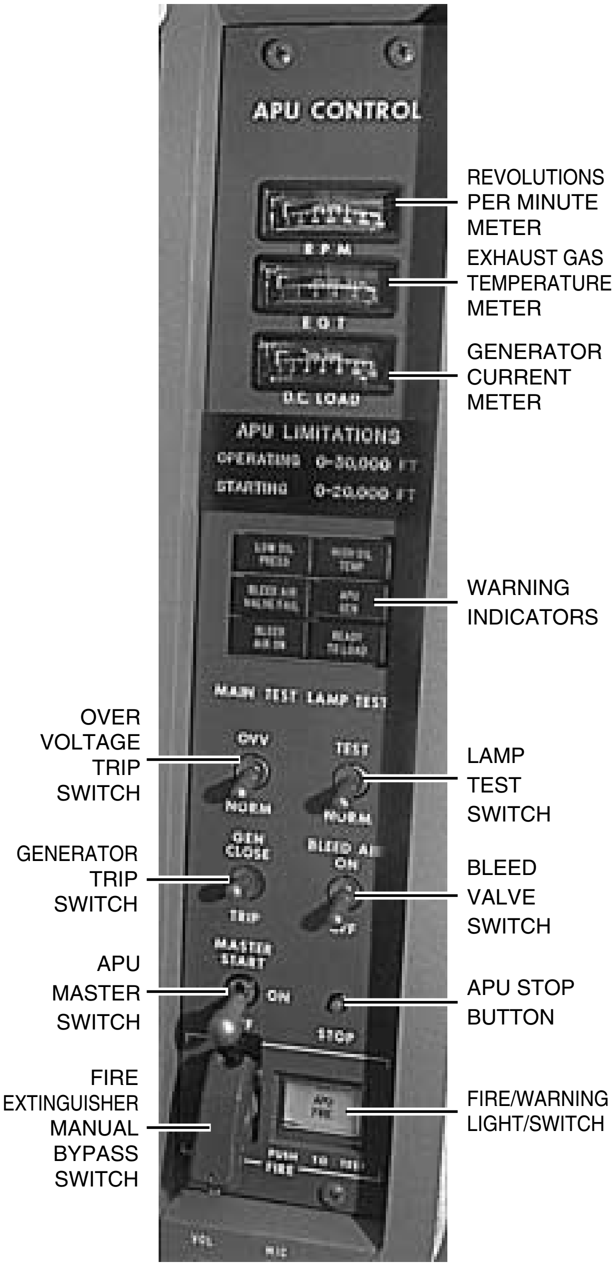


Figure 6-2. APU Control Panel

- ***STOP Button***—Initiates action to the ECU to shutdown APU.

- ***APU FIRE/PUSH TO TEST Switch***—Illuminates red when fire detectors sense an over temperature condition. Depressing switch tests fire detection circuit and silences fire bell.

- ***FIRE EXT Switch***—Initiates action to the ECU to shutdown APU. Manually discharges fire extinguisher.

- ***LOW OIL PRESS Annunciator***—Illuminates amber when oil pressure is less than 35 psig. APU shuts down.

- ***HIGH OIL TEMP Annunciator***—Illuminates amber when temperature is above 325° F. APU shuts down.

- ***BLEED AIR VALVE FAIL Annunciator***—Illuminates amber if load control valve fails to close when BLEED AIR switch is set to CLOSE.

- ***APU GEN Annunciator***—Illuminates amber when generator is off line.

- ***BLEED AIR ON Annunciator***—Illuminates green when load control valve opens.

APU Starting

The APU may be started using either the aircraft batteries or an external ground power source.

If fire extinguisher pressure is below minimums, the APU will not start.

Before Starting

1. Fire extinguisher gauge:
   Check at or above ......300 psig at –40°F
   575 psig at +70°F

2. Circuit breakers ....Set

3. APU master switch ....OFF

4. Battery switch ....ON
5. External power ....ON (if desired)
6. Main Test OVV switch ....NORM
7. TEST switch ....NORM
8. GEN switch........CLOSE (momentarily)
9. BLEED AIR switch ....OFF
10. Fire extinguishing switch ....Covered by guard
11. APU FIRE button......................Push In (check APU F IRE illuminates and fire bell sounds)

Starting

1. Aircraft LH Fuel L.P. Cock..........OPEN
2. APU MASTER switch ....ON
3. Ammeter ....Reads zero
4. APU RPM/gauge..................Reads zero
5. APU EGT gauge.........................Reads approximately ambient
6. Aircraft LH Boost Pump ....ON (check FUEL 1 LO PRESS annunciator light extinguishes)
7. APU LOW OIL PRESS........Illuminated
8. APU GEN ....Illuminated
9. Remaining APU annunciators ....Not illuminated
10. APU MASTER switch ....START (momentarily)

    Check:

    a) APU RPM gauge shows rotation.

    b) APU EGT gauge shows rise within 10 seconds after rotation.
11. EGT gauge ....Monitor start cycle
12. RPM gauge ....Monitor start cycle

    Check:

    a) Below 50% RPM..........Max 760°C to 870° C for 15 sec
    870° C to 974° C for 10 sec

    b) Below 95% RPM ....Loss of acceleration (hung start)
    Do not exceed 10 secs
13. APU LOW OIL PRESS......Extinguishes
14. APU GEN ....Extinguishes
15. READY TO LOAD ....Illuminates
16. Ammeter........................Indicates Load
17. RPM gauge ....Check 100%
18. EGT gauge ....Check within limits

CAUTION

For unattended operation, the aircraft battery power is left on line to enable discharge of the fire extinguisher bottle in event of an auto shutdown caused by the fire detector system.

Systems Operation – Air Conditioning and Electrical Power Extraction

When desired the APU can be used to provide bleed air for the environmental control system (E.C.S) and DC generator electrical power.

The system can provide these functions either simultaneously or singly as selected. The following instructions denote selection of systems desired.

Simultaneous Bleed Air and Electrical System Operation

1. APU ....Start (check operating within limits)
2. BLEED AIR Switch ....ON
   Check:
   a) BLEED AIR ON ....Illuminates
   b) BLEED AIR VALVE FAIL...........Not illuminated

CAUTION

If the BLEED AIR VALVE FAIL illuminates amber, do not turn on main engine bleed air until the load APU control valve malfunction is corrected.

3. Cabin Condition Controls ....Set as desired
4. RPM gauge .... Check 100%
5. EGT gauge ....Check within limits
6. Ammeter ....Check loading within limits

Bleed Air Only (Remove Electrical Load)

1. GEN switch ....OFF
2. APU GEN ....Illuminates
3. Ammeter ....Check reads zero

Electrical Load Only (Remove Bleed Load)

1. BLEED AIR switch ....OFF
2. BLEED AIR ON ....Extinguished
3. READY TO LOAD.............Illuminated
4. GEN switch ....CLOSE
5. APU GEN ....Extinguished

Shutdown

NOTE

It is not necessary to remove any load from the APU to shutdown.

1. STOP switch ....Depress momentarily
2. RPM gauge ......Check decrease in RPM
3. EGT gauge....................Check decrease in temperature
4. MASTER switch ....OFF when RPM indicates zero

Abnormal Conditions

Automatic Shutdown

a. APU fire
b. Low oil pressure
c. High oil temperature
d. DC power interrupt to ECU
e. Overcurrent to ECU
f. Loss of EGT signal
g. High EGT above the red line
h. Overspeed in excess of 110%
i. Loss of speed signal

In case of automatic shutdown (for fire see Emergency Procedures):

1. MASTER switch ....OFF

WARNING

Do not operate APU until cause of shutdown has been determined and corrected.

Do not operate the airplane if fire is suspected as the cause of automatic shutdown until damage has been assessed.

NOTE

Do not return the MASTER switch to ON position since this will reset the ECU and eradicate the BITE indicators necessary to determine the cause of shutdown.

After shutdown all annunciator lights will extinguish unless caused by fire in which case the APU FIRE will remain illuminated until temperatures drop below alarm point.

Failure to Start

If the APU fails to start within ten seconds of activating the MASTER switch to start:

1. STOP switch ....Depress
2. MASTER switch ....OFF

CAUTION

Wait 3 minutes to allow fuel to drain off before attempting another start.

Emergency Procedures

APU Fire

In the event of a fire in the APU enclosure the APU FIRE button will indicate red and an audible warning bell will sound. If the APU is operating, this will trigger an automatic shutdown of the APU. After five seconds of fire warning the fire extinguisher will automatically discharge.

Manual shutdown of the APU and manual discharge of the fire extinguishing system can be performed as follows:

1. STOP switch ....Depress momentarily
2. MASTER switch ....OFF
3. Wait 5 seconds.
4. FIRE EXTINGUISHER switch ....Lift guard
   Select discharge

WARNING

Do not operate airplane or APU after a suspected fire until possible damage has been assessed.

NOTE

Do not return the MASTER switch to ON.

The audible warning horn may be silenced after temperature in APU enclosure has returned to normal by pushing the reset switch located in the aft compartment where the APU is installed.

If horn sounds in flight, it may be muted by depressing APU FIRE switch, and will cease sound if temperature condition draws below alarm point.

Main DC Generator Failure

Do not start APU if both main engine generators have failed. If both engine generators have failed and the APU is already operating, the APU may be left on provided the APU generator is operating. Follow emergency procedures in the basic *Airplane Flight Manual* for two main generators inoperative.

WARNING

A failed APU start with two inoperative main generators may deplete the aircraft batteries and leave the aircraft without electric power.

TURBOMACH APU (T62T-40)

GENERAL

The Hawker 800 XP series may be equipped with an APU (Figure 6-3) manufactured by Solar Turbines International. This APU is designated T62T-40. The turbomach APU is certificated for both on-ground and in-flight (for emergency use only) operation. Flight operation is limited to a maximum altitude of 15,000 feet (emergency only).

MAJOR SECTIONS

The Solar APU (Figure 6-4) is a small gas turbine engine consisting of six major sections:

1. Air inlet
2. Compressor
3. Combustor
4. Turbine
5. Exhaust
6. Reduction drive assembly

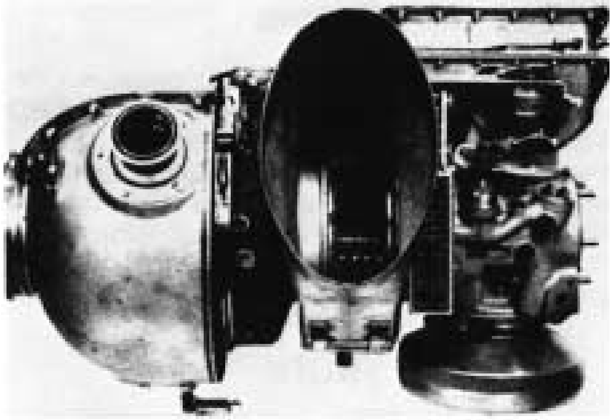


Figure 6-3. Turbomach APU (Typical)

Air Inlet

The air inlet is a screened, circular opening in the waist of the APU, and it provides the main air inlet to the compressor.

Compressor

The compressor is a single-stage centrifugal impeller which draws air through the inlet, compresses the air, and directs the airflow for cooling combustion and bleed-air extraction.

Combustor

The combustor consists of a shrouded, reverse-flow annular chamber supplied with a specific air volume from the compressor. Fuel is added to the combustion air by a start fuel injector and three main fuel injectors, each controlled by a solenoid valve. In addition, a maximum fuel solenoid valve increases fuel flow to the main fuel injectors during high load periods.

Ignition is supplied by a single igniter plug.

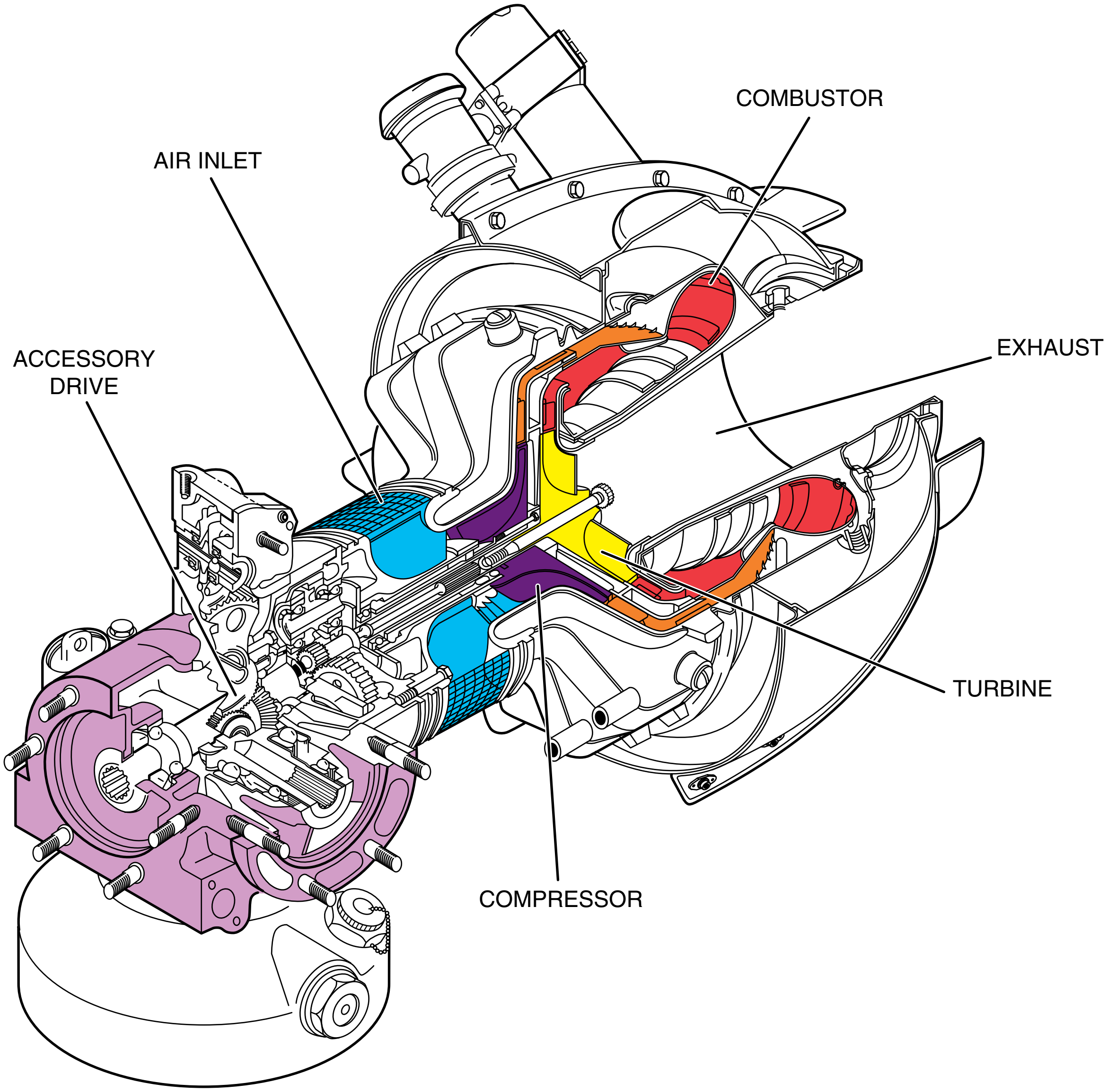


Figure 6-4. Major Sections (Turbomach APU)

Turbine

The single-stage, radial-flow turbine is mounted piggyback on the compressor. The turbine extracts almost all the energy from the expanding combustion gases and uses the energy to drive the compressor and the accessory drive. The flow of spent gases from the turbine is reversed to flow aft to the exhaust.

Exhaust

The exhaust consists of an exhaust duct and an exhaust pipe to direct the spent gases to the atmosphere through an outlet on the right side of the rear fuselage above the right engine nacelle.

Reduction Drive Assembly

The reduction drive assembly is a small planetary gear assembly driven by the turbine shaft. The planetary system reduces the high turbine rpm to the values required for the various accessories which consist of the following:

- Oil pump
- Fuel pump and fuel controller
- DC starter-generator
- Tach generator

OIL SYSTEM

General

The APU oil system is a self-contained, fully automatic system which provides for cooling and lubrication of the rotor bearings and the bearings and gears in the reduction drive assembly.

FUEL SYSTEM

General

The APU fuel system is a fully automatic system with no operator controls or adjustments, The fuel system functions to control the APU during starting and acceleration, and provides for a nearly constant rpm under all operating conditions.

IGNITION SYSTEM

General

The APU ignition system is a high-energy capacitor discharge system. The ignition exciter is mounted on the APU and supplied with DC power from the distribution system. A high-tension lead connects the exciter to a single igniter plug in the combustor. Ignition is operational from the initiation of the startup to 90% rpm when it is terminated by a speed-sensing switch.

Electronic Sequence Unit (ESU)

The ESU controls startup and monitors the APU while it is operating. Any malfunction of the APU will result in the ESU shutting it down.

CONTROL AND INDICATIONS

General

Control, indication, and warning for the APU (Figure 6-5) are all provided on a single panel located on the side face of the bulkhead behind the pilot's seat. The panel contains the following controls and indicators.

APU MASTER Switch

The APU MASTER switch is a two-position, ON–OFF switch. When the switch is on, electrical power is available to the APU control panel.

TURBINE START Switch

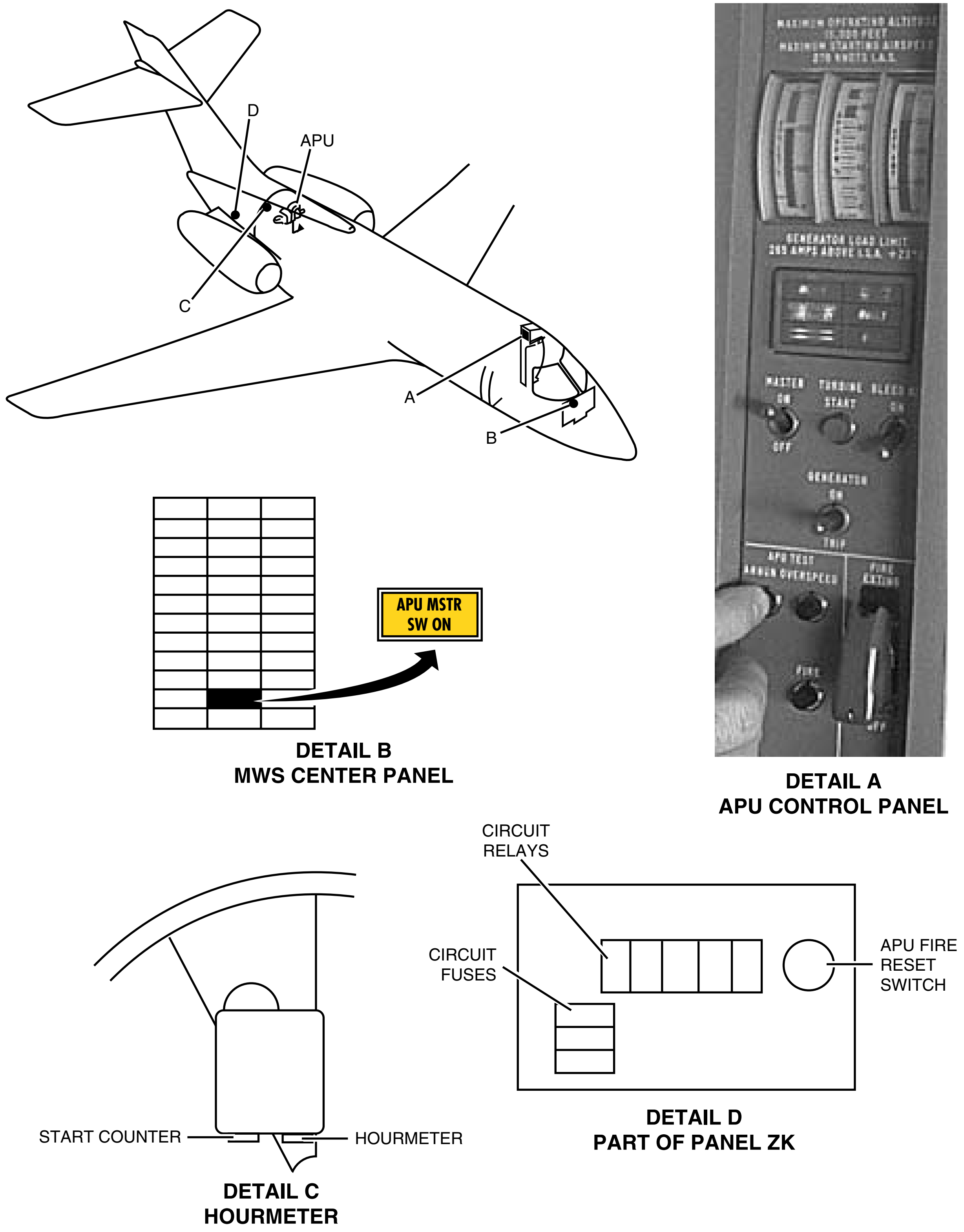
The TURBINE START switch is a momentary, pushbutton switch. When pushed, the switch will initiate the automatic start sequence of the APU.

BLEED AIR Switch

The BLEED AIR switch is a two-position, ON–OFF toggle switch. The function of the BLEED AIR switch is to control the electropneumatic bleed valve that supplied APU bleed air to the airplane's environmental system when the airplane is on the ground. For more information on APU bleed air, see Chapter 9, "Pneumatics" and Chapter 11, "Air Conditioning."

NOTE

With the Turbomach APU, APU bleed air is available in flight.



DETAIL B
MWS CENTER PANEL

DETAIL A
APU CONTROL PANEL

DETAIL D
PART OF PANEL ZK

DETAIL C
HOURMETER

Figure 6-5. APU Control Panel (Turbomach APU)

GENERATOR Switch

The GENERATOR switch is a three-position, ON–OFF–TRIP switch that provides for control of the APU starter-generator. For more information on the APU generator, see Chapter 2, "Electrical Power Systems."

APU Test Panel

General

An APU test panel forms part of the APU control panel and contains the following switches.

OVERSPEED Test Switch

When pushed, the switch tests the overspeed detection system and turns on the amber FAULT light on the APU annunciator panel. The switch will also shut down the Turbomach T62T-40 APU.

ANNUNciator Test Switch

The ANNUNciator test switch is a two-position, momentary pushbutton switch. When pushed, all lights on the APU annunciator panel will come on as will any optional lights on the master warning lights panel.

FIRE Test Light

This two-position, momentary switch, when pushed, will turn on the red FIRE light, the airplane's master warning flashing lights, the fire bell, and the APU warning light on the MWS panel.

The APU annunciator panel also includes a green light marked "GEN ON DC BUS." This light will illuminate to indicate that the APU starter-generator is supplying DC power to the distribution system.

The amber GENERATOR OVERHEAT light will come on if the operating temperature exceeds a preset value.

The red FIRE light will come on if a high temperature or fire exists in the APU area. Fire and/or overheat will automatically close the bleed-air valve.

An amber FAULT light will come on for the following conditions:

- Low oil pressure
- High oil temperature
- High EGT
- Overspeed

EGT Gage

The EGT gage is calibrated in degrees Fahrenheit and receives inputs from a thermocouple probe mounted in the exhaust duct.

APU RPM

The APU rpm gage receives input from the tach generator driven by the APU reduction drive assembly. The APU rpm is calibrated in percent of design maximum.

APU Ammeter

The APU ammeter indicates the output of the APU generator to the DC bus system.

PROTECTION

General

An APU protection system provides for automatic APU shutdown if (1) the APU EGT exceeds a predetermined value, (2) the APU oil pressure drops to a low value, (3) the APU rpm exceeds a predetermined value, or (4) an APU fire is detected.

NOTE

If the APU is shut down for fire warning, restart cannot be accomplished until the APU fire reset switch, located on panel ZK, has been reset.

APU STARTING

General

The APU should not be started until the prestart checklist is completed and all discrepancies are corrected. When ready to start the APU, push the TURBINE START switch momentarily and then release the switch. Monitor the APU rpm and EGT until both stabilize at the termination of the start sequence.

CAUTION

> If EGT exceeds the red line value or if rpm hangs or decreases during the start sequence, turn off the APU MASTER switch and do not attempt another start until 30 seconds after the APU rpm reaches 0%.

Following a normal start, if the APU generator ammeter does not show a lead, momentarily move the GENERATOR switch to ON, then release the switch and monitor the EGT gage.

NOTE

> EGT will increase when the APU generator is turned on.
>
> A start inhibit facility prevents APU starting while on external power and the EXT BATT CHG switch, located on the pilots' roof panel, is set to ON.
>
> The APU generator will not connect to the bus system if the external power generator is supplying the bus system.
>
> Before turning the APU BLEED AIR switch on, the APU rpm will indicate approximately 102%.

If bleed-air extraction is required for the environmental system, turn on the APU BLEED AIR switch. EGT will increase and rpm will decrease.

NOTE

> APU EGT is an indirect function of the cabin temperature controller. If the controller is at the maximum heat position, the APU EGT will be high. Remember, APU EGT will rise in proportion to the bleed-air valve opening and the DC generator lead.

ELECTRICAL POWER SOURCES

GARRETT AND TURBOMACH APU

PE bus bar

- APU master switch
- Fire warning system
- Fire extinguisher

QUESTIONS

1. When the APU generator is turned on:

A. RPM and EGT will decrease.
B. RPM and EGT will increase.
C. EGT will decrease and RPM will increase.
D. RPM will decrease and EGT will increase.

2. If EGT increases or rpm hangs or decreases during a start:

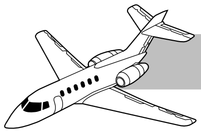
A. Turn off the BLEED AIR switch.
B. Turn off the APU MASTER switch.
C. Discharge the APU fire bottle.
D. Trip the generator.

3. Do ***not*** operate APU bleed air and main engine bleed air simultaneously for longer than:

A. 30 seconds
B. 1 minute
C. 3 minutes
D. 6 minutes

4. Normal APU shutdown is accomplished by:

A. Turning off battery switch
B. Turning off APU master switch
C. Depressing stop switch
D. None of the above



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POWERPLANT

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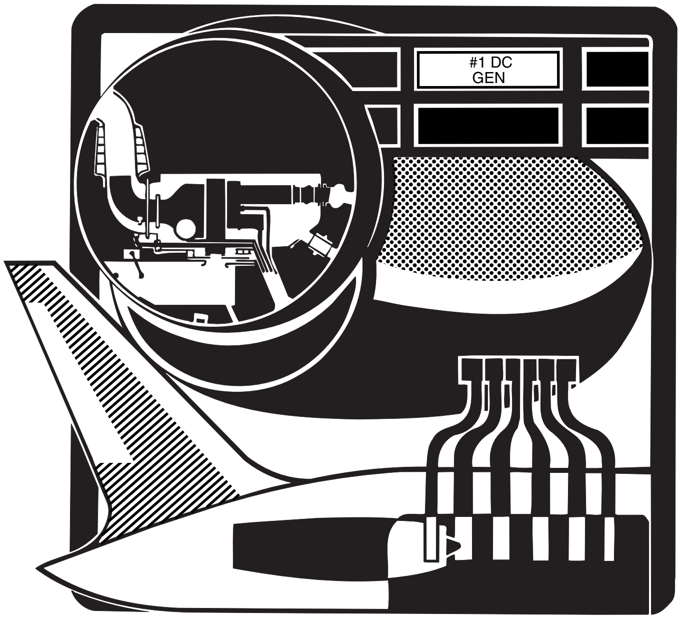
POWERPLANT



INTRODUCTION

This chapter describes the powerplants installed on the Hawker-800 XP series airplanes. In addition to the engine, the chapter also describes such related systems as oil, fuel, ignition, thrust reverser, engine instrumentation, engine control, engine starting, engine synchronization, and automatic performance reserve. Pertinent limitations affecting the engine and related systems are included at the end of the chapter.

GENERAL

Thrust is supplied by two aft-fuselage, pylon-mounted engines manufactured by Garrett Turbine Engine Company at Phoenix, Arizona, a division of Allied Signal. The engines are designated TFE731-5BR-1H.

The TFE series engines are lightweight, twin-spool turbofans. The modular design concept of the engines facilitates maintenance and reduces airplane downtime.

Each engine develops 4,660 pounds of thrust, static, at sea level in temperatures up to 73° F.

Each engine includes a fully automatic lubrication system, engine fuel and ignition systems, two-zone fire and overheat detection, single-zone fire extinguishing systems, and engine anti-ice systems.

ENGINES

The engines (Figure 7-1) are enclosed in close-cowled streamlined nacelles. The cowls (two per engine) can be completely removed to expose the engine and all of its components.

Figure 7-2 illustrates a typical 731 series engine.

MAJOR COMPONENTS

The engine consists of five major components:

- Fan
- Low Pressure (LP) Spool
- High Pressure (HP) Spool
- Annular Combustion Chamber
- Transfer and Accessory Gearboxes.

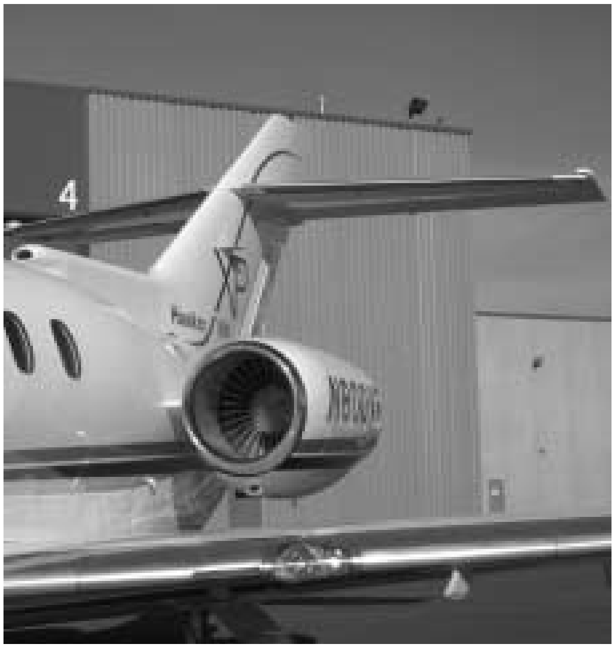


Figure 7-1. View of Aft-Fuselage-Mounted Engine

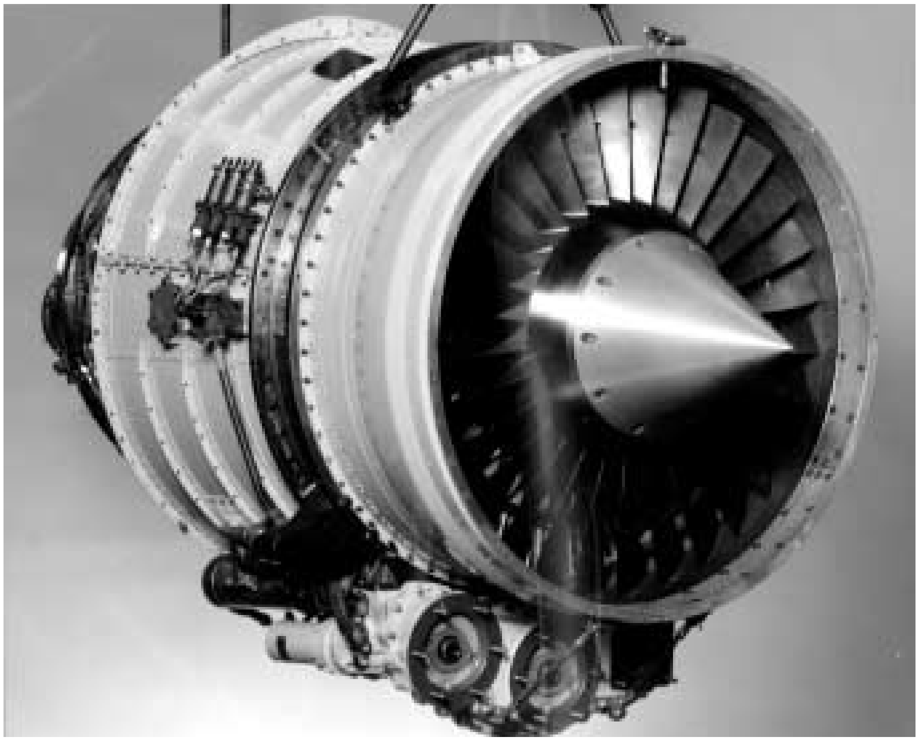


Figure 7-2. Typical 731 Series Engine

DESCRIPTION

The engine is a two-spool-transonic-stage-compressor, front fan jet engine. It is a light weight modular design for ease of maintenance. The simplicity of the design eliminates the need for variable geometry inlet guide vanes. This minimizes the weight of the engine, the possibility of the inlet vanes icing up is reduced and the noise is also reduced. Use of a reverse flow combustion chamber reduces the overall length of the engine and provides a cool skin concept for the external surfaces of the turbine section. (See Figure 7-3.)

FAN

The fan is an axial flow unit that moves large quantities of air into the bypass and core inlets. The bypass section consists of the fan spinner support, fan rotor assembly, fan bypass stator, fan duct assembly and the bypass fan support and shaft section. The fan is driven by the low pressure (N_1) spool through the planetary gear section.

Compressor Section

Air enters the engine through the air inlet section located immediately aft of the fan bypass section.

From the air duct it enters the compressor section at the LP compressor where it is compressed and forced through the interstage diffuser assembly to the HP compressor. Air is further compressed by the high pressure compressor and is discharged into the combustion chamber.

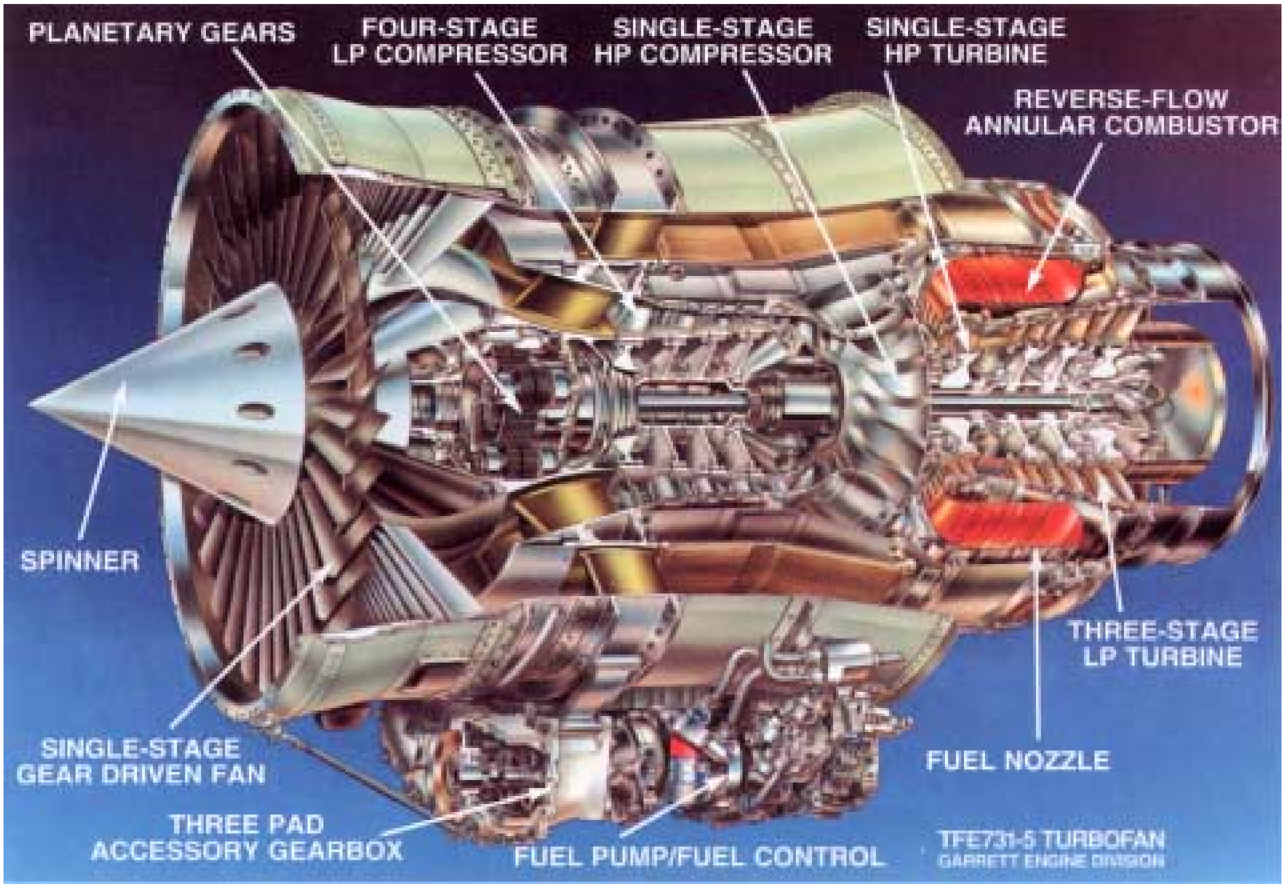


Figure 7-3. Engine Cutaway

Low Pressure Spool (N_1)

The LP (N_1) spool consists of a four stage, low pressure, axial flow compressor and a three stage, low pressure turbine. Both the compressor and the turbines are mounted on a common shaft.

NOTE

A stage is a paired rotor and stator.

Each stage of the axial flow compressor utilizes rotating compressor blades to accelerate the air, followed by static stator blades which decelerate the air, converting kinetic energy into pressure. This provides a steady rise in pressure through the compressor stages, without significant change to overall velocity.

High Pressure Spool (N_2)

The high pressure spool (N_2) consists of a single stage centrifugal compressor driven by a single stage turbine through an outer concentric shaft.

The centrifugal compressor consists of an impellar (rotor), a diffuser and a compressor manifold. As in axial flow compressors, air is picked up and accelerated outwards towards the diffuser. When the accelerating air reaches the diffuser its velocity is reduced, converting kinetic energy into pressure.

The high pressure spool also drives the accessory gearbox through a tower shaft and transfer gear reduction system.

Combustion Chamber

The compressed air flows into a single reverse flow annular combustion chamber in the turbine section where it is mixed with atomized fuel supplied by twelve duplex fuel nozzles. The twelve duplex fuel nozzles consists of primary nozzles used for starting and secondary nozzles used in conjunction with the primary nozzles for all other phases of engine operations.

The fuel-air mixture is ignited by the two igniter plugs located at the six and seven o'clock positions within the combustion chamber. After the ignition cuts-out, combustion is self sustaining. The resultant combustion gases are then directed to the turbine by the transition liner.

The hot gases pass through both the high and low pressure turbines, driving both rotating compressor assemblies and then exiting through the exhaust nozzles with the bypassed air.

Turbine Section

The turbine section contains four (one high pressure, three low pressure) axial flow turbine wheels and four stator assemblies. On leaving the turbine, the exhaust gases enter a mixer compound-thrust-nozzle system, where they mix with the bypass air before discharging through a convergent-divergent nozzle.

The high pressure turbine rotor assembly is air cooled to allow an increased turbine inlet temperature.

A lobed mixer nozzle fitted to the core engine, mixes hot core exhaust gas with the bypass air before discharging the mixed flow through the convergent-divergent nozzle. Mixing the hot and cold flows results in more efficient use of the thermal energy in the hot core flow and also benefits smoke and noise reduction.

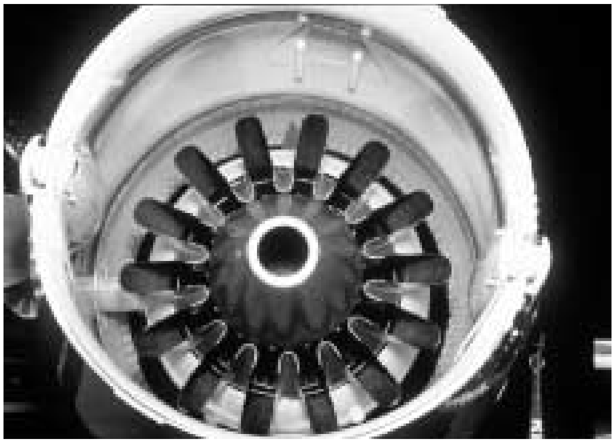


Figure 7-3A. Engine Exhaust

Turbine Temperature

Ten thermocouples (two pairs of five thermocouples connected in parallel to create an averaging circuit) are located in the gas path between the high pressure turbine and the first stage of the low pressure turbine. These thermocouples measure the interstage turbine temperature (ITT) and send signals to the cockpit indicator and to the fuel computer.

An indicator for each engine is located on the center instrument panel and this displays the average interstage turbine temperature (ITT). The indicator displays ITT from 0° C to 1,200° C in 50° C increments with colored bands to indicate different temperature ranges. An off flag indicates power loss to the indicator.

To facilitate the accurate setting of climb power, a green light mounted in the corresponding ITT indicator will illuminate when PLA equals 105 ±2 degrees. The light will remain on until PLA is moved above or below the specified range.

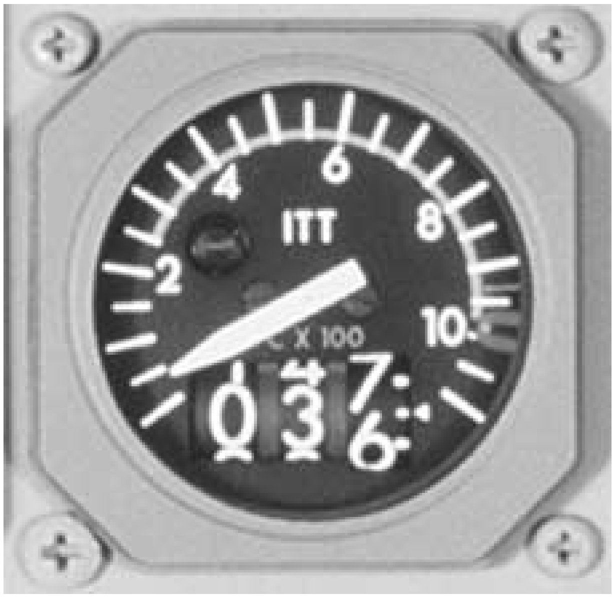


Figure 7-3B. ITT Gage

Accessory Drive

An accessory drive gearbox and transfer gearbox are driven from the high pressure (N_2) spool. The transfer gearbox is driven by a vertical shaft and in turn drives the accessory gearbox through a horizontal gearshaft. The accessory drive gearbox provides shaft power for aircraft accessories (hydraulic pump, starter/generator and alternator) which are mounted on the forward face of the accessory gearbox. The fuel pump, fuel control unit and oil pump are all mounted on the rear face of the accessory gearbox.

OPERATION

When the engine is operating (Figure 7-4), the single-stage fan draws air in through the nacelle inlet duct. The outer diameter of the fan accelerates a moderately large air mass at a low velocity into the full-length bypass duct. At the same time, the inner diameter of the fan accelerates an air mass into the engine core. The pressure of this air is increased by the LP compressor and directed to the HP compressor where the air pressure is further increased and ducted aft to the combustor.

A precise amount of this air enters the reverse-flow combustor where fuel is injected by the twelve spray nozzles. The mixture is initially ignited by the two igniter plugs and expanded through the turbine. The HP turbine extracts enough energy to drive the HP compressor and the transfer and accessory gears. The LP turbine extracts enough energy to drive the LP compressor, the planetary gear, and the fan.

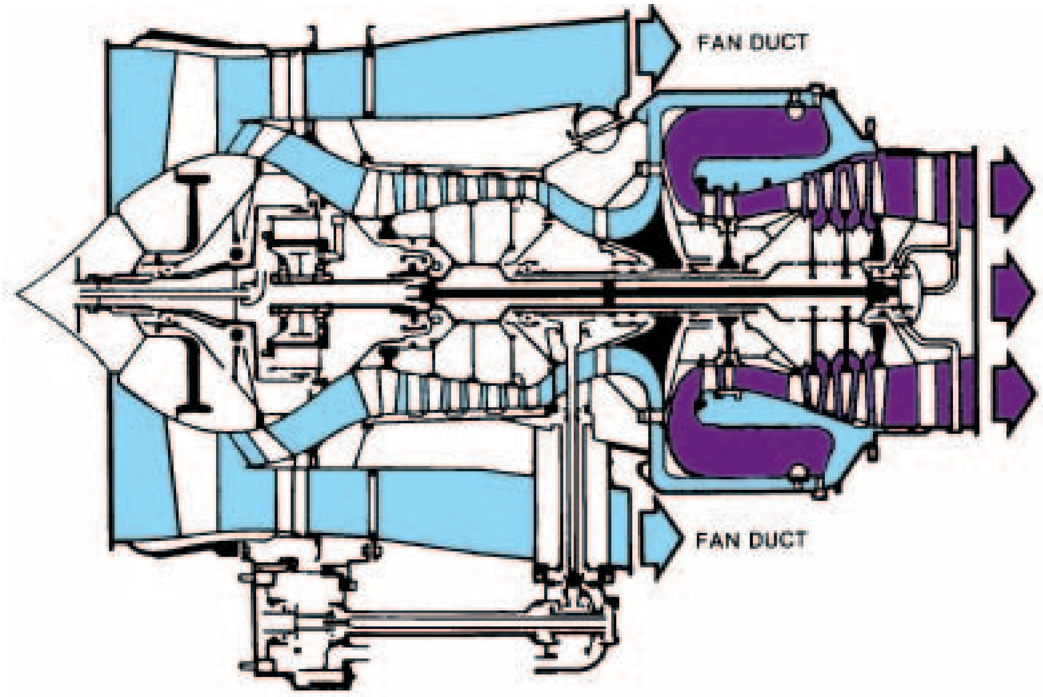


Figure 7-4. Gas Flow

The remaining gas energy is accelerated aft through the exhaust pipe and joins the fan airflow from the bypass duct to provide the total direct thrust.

ENGINE SYSTEMS

Powerplant systems include:

- Engine Oil
- Ignition
- Fuel Control
- Power Controls
- Air Bleed
- Automatic Performance Reserve (APR)
- Thrust Reverser.

ENGINE OIL

Oil under pressure lubricates the engine bearings and the transfer, accessory and planetary gearboxes. The system consists of (see Figure 7-5):

- Oil Tank and Sight Gage
- Oil Pump
- Oil Filter and Bypass Valve
- Air/Oil Cooler and Bypass Valve
- Fuel to Oil Cooler
- Oil to Fuel Cooler
- Breather Pressurizing Valve
- Pressure and Temperature Transmitters and Indicators.

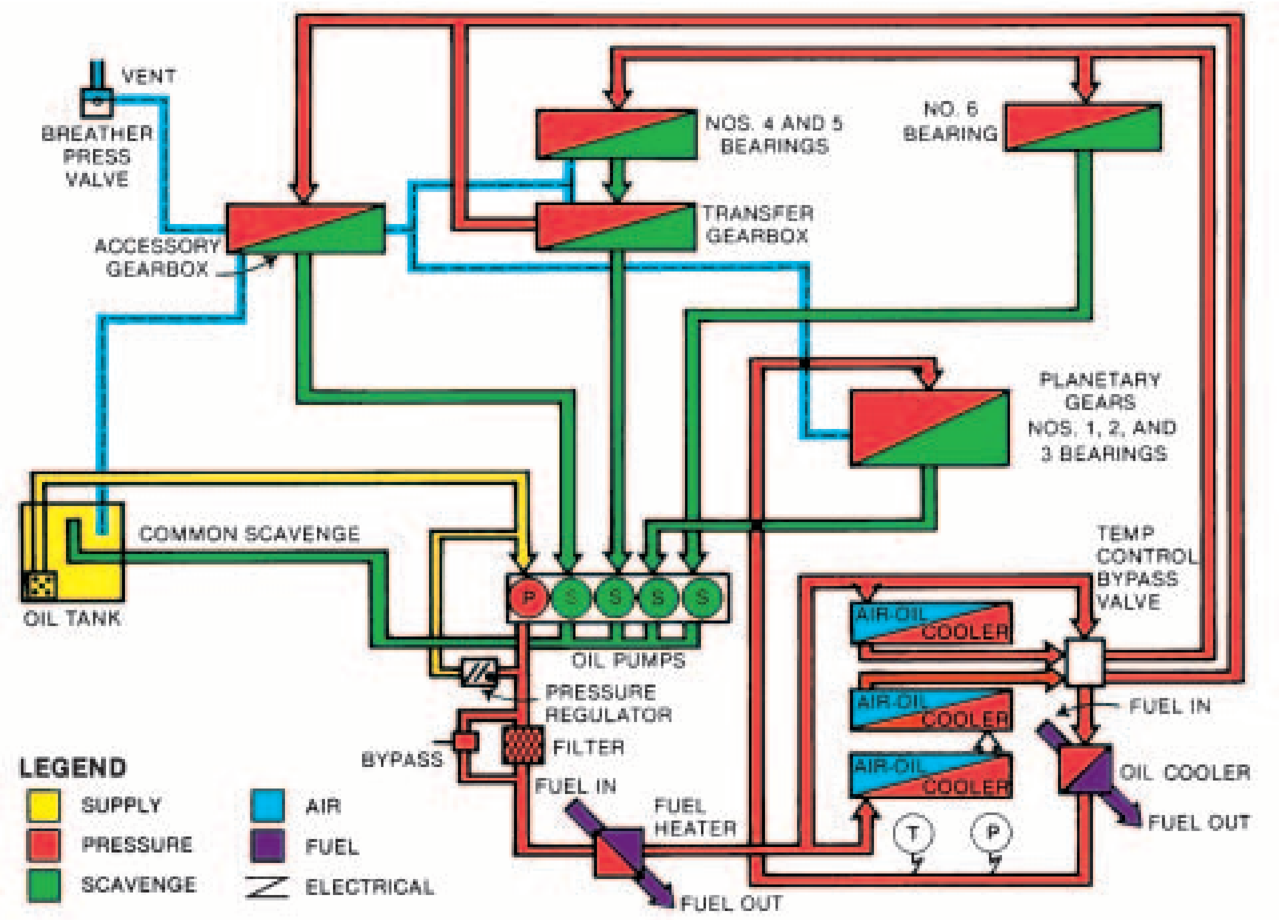


Figure 7-5. Oil System

Rotation of the engine-driven oil pump draws oil from the reservoir. Oil under pressure flows through a pressure regulator, filter and temperature control components to the engine bearings, the transfer gearbox, accessory gearbox and the front fan planetary gear assembly.

The reservoir has a liquid level sight gage and a filler cap on the right side of the tank. A filler tube and cap are located on the left side of the tank which allows for oil tank replenishing when access to the right side is restricted.

Oil Pump Assembly

An oil pump assembly is located on the accessory drive gearbox. It contains a single oil pressure pump and four scavenge pumps.

The pressure pump draws oil from the reservoir and supplies sufficient pressure to force the oil through the engine components that require lubricating.

The scavenge pumps collect oil from the planetary gear assembly and the forward engine bearings, the aft engine bearings, the transfer gearbox and the mid engine bearings, and the accessory drive gearbox. A common discharge line connects the four scavenge pumps to the engine oil reservoir.

An adjustable pressure regulator in the pumps helps to provide a constant oil pressure by compensating for changes in the aircraft altitude.

Oil Filter

A filter is provided to remove impurities from the oil. The oil filter consists of a disposable element enclosed in a metal housing on the right side of the accessory drive gearbox.

Oil Tank

A 1.65 U.S. gallon capacity oil reservoir is located on the right side of the engine fan bypass housing.

Engine protection against filter clogging is provided by an oil filter bypass indicator valve located adjacent to the oil filter. The valve opens when the pressure drop across the filter is excessive to bypass lubricating oil around the filter.

An integral differential (ΔP) pressure indicator on the valve visually flags a clogged filter condition before bypassing occurs. A thermal lockout device on the ΔP indicator prevents actuation under cold oil conditions although the bypass valve will bypass oil under these conditions.

Fuel Heater

An externally mounted fuel heater is located on the left side of the engine. The fuel heater provides oil-to-fuel heat exchanging to prevent ice formation in the fuel system from clogging the fuel filter and any other components.

Fuel flow through the fuel heater is thermostatically controlled to provide the optimum operating temperature. Excess oil pressure with cold oil is prevented by a pressure bypass valve.

Air/Oil Cooler

After oil leaves the fuel heater, it passes through the air/oil coolers in the engine bypass duct. The air/oil cooler consists of three segments: one half segments and two quarter segments.

Each segment is a finned unit with oil lines running through it. Together the three segments form the inner surface the fan duct. A temperature controlled integral bypass valve directs oil that is hotter than 65° C through the three segment of the air/oil cooler. Air flowing through the duct cools the oil that passes through the cooler. Below this temperature, the valves bypass the oil around the air/oil cooler.

If the heat exchangers become obstructed, the temperature control valve bypasses the oil around them.

After the oil leaves the air/oil coolers, the flow splits. Part of the oil flows to the engine bearing sumps (HP rotor shaft), the transfer gearbox assembly and the accessory gearbox. The remaining oil flows through the oil temperature regulator (fuel/oil cooler) and then on to the planetary gear assembly.

Fuel/Oil Cooler

The fuel/oil cooler (oil temperature regulator) uses aircraft fuel to maintain the oil at a constant temperature.

The unit consists of a temperature control valve and a heat exchanger. Whatever the oil temperature, fuel constantly flows through the unit: oil only flows through the unit if it is above a set temperature. If the oil temperature exceeds 99° C, the control valves open to rout the oil through the cooler.

From the fuel/oil cooler, oil then lubricates the fan shaft bearings and the front LP spool bearings. After travelling to all the main sump areas, oil then drains by gravity to the lowest point of each sump. The scavenge pumps then draw the oil back to the engine oil reservoir.

Oil Venting

Vent lines interconnect the oil sumps to the oil tank assembly and the breather pressurizing valve. The breather pressurizing valve provides an ambient vent for the oil system at low altitudes and at high altitude increases the internal engine vent and tank pressure to ensure proper oil pump operation.

Oil Pressure and Temperature Indications

Indicators scaled from 0 to 70 psi display the oil pressure for each engine. A pressure transmitter in the oil supply manifold for each engine's planetary gear assembly supplies electrical signals to the two oil pressure indicators on the center instrument panel.

Temperature bulbs in each engine's oil supply manifold are connected electrically to two oil temperature indicators on the center instrument panel. Each indicator has a single pointer that moves over a scale marked from 0° C to 170° C increments (see Figure 7-6).

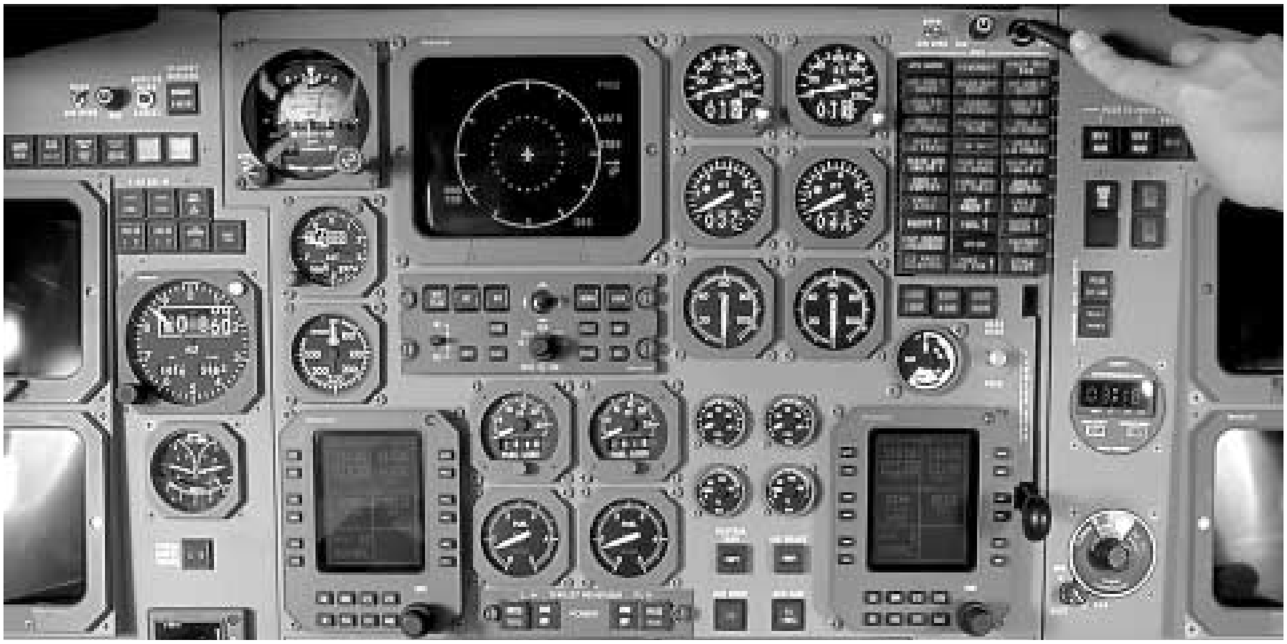


Figure 7-6. Center Instrument Panel

Oil Warning Annunciators

Pressure switches in each engine's oil supply line operate the red OIL 1 LO PRESS and OIL 2 LO PRESS annunciators on the MWS (see Figure 7-6).

Normally, engine oil pressure holds the switches open. If the pressure drops below 23 psi, the switch will close to complete a circuit which will cause the respective annunciator to become lit. Once the pressure exceeds 25 psi the switch will open and break the circuit which will cause the annunciator to become unlit.

IGNITION

Each engine has an independent ignition system that consists of:

- Ignition Unit
- Igniter Leads
- Igniter Plugs
- ENG IGNITION Switches
- IGN ON Annunciators.

Ignition Unit

An ignition unit on the upper left side of the fan bypass housing is a high voltage, capacitor discharge, radio noise-suppressed, intermittent sparking type unit that use a 10 to 30 VDC power supply. The ignition system receives power from the PE busbar. Each unit provides separate and independent outputs of 18,000 to 24,000 volts to the igniter plugs.

During the engine start cycle, a microswitch on each HP fuel/lever provides ignition unit activation. Once the engine reaches self-sustaining speed, the relays deenergize to remove power from the ignition units.

Manual operation of the ignition unit is through the ENG IGNITION switch in the ON position. If required, the unit can be operated continuously.

Igniter Plugs

The igniter plugs, on the annular combustion chamber at the six and seven o'clock position, operate independently of each other. Each receives power from the ignition unit through separate high-tension leads. Each plug fires at a rate of approximately two sparks per second when triggered by the ignition unit.

Ignition Switches

Each engine has a two-position (ON/OFF) ENG IGNITION switch on the cockpit overhead panel (see Figure 7-7).

In the ON position, the switch will cause the IGN ON annunciator to become lit and will supply the ignition unit with 28 VDC from the PE busbar through the No. 2 start auxiliary relay. The annunciator only indicates that power is available to the ignition unit. Verification of the igniter firing requires the ENG IGNITION switch to be turned to the ON position and listening for two distinct snaps in the engine area.

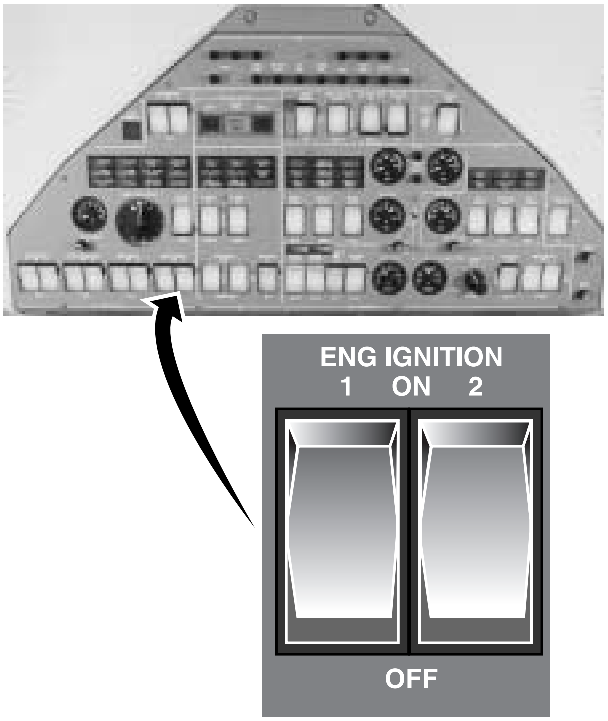


Figure 7-7. Engine Ignition Switches

FUEL CONTROL

INTRODUCTION

The engine fuel control system consists of:

- Fuel Pump Assembly
- Hydromechanical Fuel Control Unit (FCU)
- Digital Electronic Engine Control (DEEC)
- Fuel Flow Divider Assembly
- Fuel Atomizers.

From the aircraft fuel system, the fuel control system pumps, filters, meters and atomizes the fuel before the ignition system ignites it to produce thrust.

FUEL PUMP

An engine-driven fuel pump on the rear of the accessory gearbox provides high pressure fuel to the fuel control system. The pump assembly consists of:

- Booster Pump Element
- Fuel Filter
- Filter Bypass Valve
- High Pressure Pump Element
- Relief Valve.

Attached to the rear of the pump is the fuel control unit. See Figure 7-8.

Anti-ice Valve

An anti-icing valve is provided within the fuel pump assembly to mix warm fuel from the fuel heater with the discharge flow of the booster pump to prevent icing of the fuel filter element.

Filter Bypass

The filter bypass valve allows fuel to bypass the filter if an excessive pressure drop across the filter occurs.

An electrical pressure switch will cause the respective annunciator ENG # FUEL to become lit when an excessive differential pressure condition exists before bypassing occurs or excessive fuel temperature.

Filter Clogged

If the filter starts to become clogged, the following events occur:

- At 6 to 8 psi Δp the amber annunciator will be lit on the MWS.
- At 9 to 12 psi Δp the filter bypass valve opens to deliver fuel to the high pressure pump.

The annunciator will remain lit for as long as the fuel filter remains clogged.

FUEL CONTROL UNIT

The fuel pump-driven fuel control unit (FCU) on the rear of the fuel pump contains:

- Fuel Metering Section
- Power Lever Input Pot
- Shutoff Valve
- Outlet Pressurizing Valve
- Ultimate Overspeed Solenoid
- Mechanical Governor (N_2)

The mechanical governor functions in two modes: as an overspeed governor for the HP rotor if the fuel computer is operative; or, as a hydromechanical control when the fuel computer is inoperative.

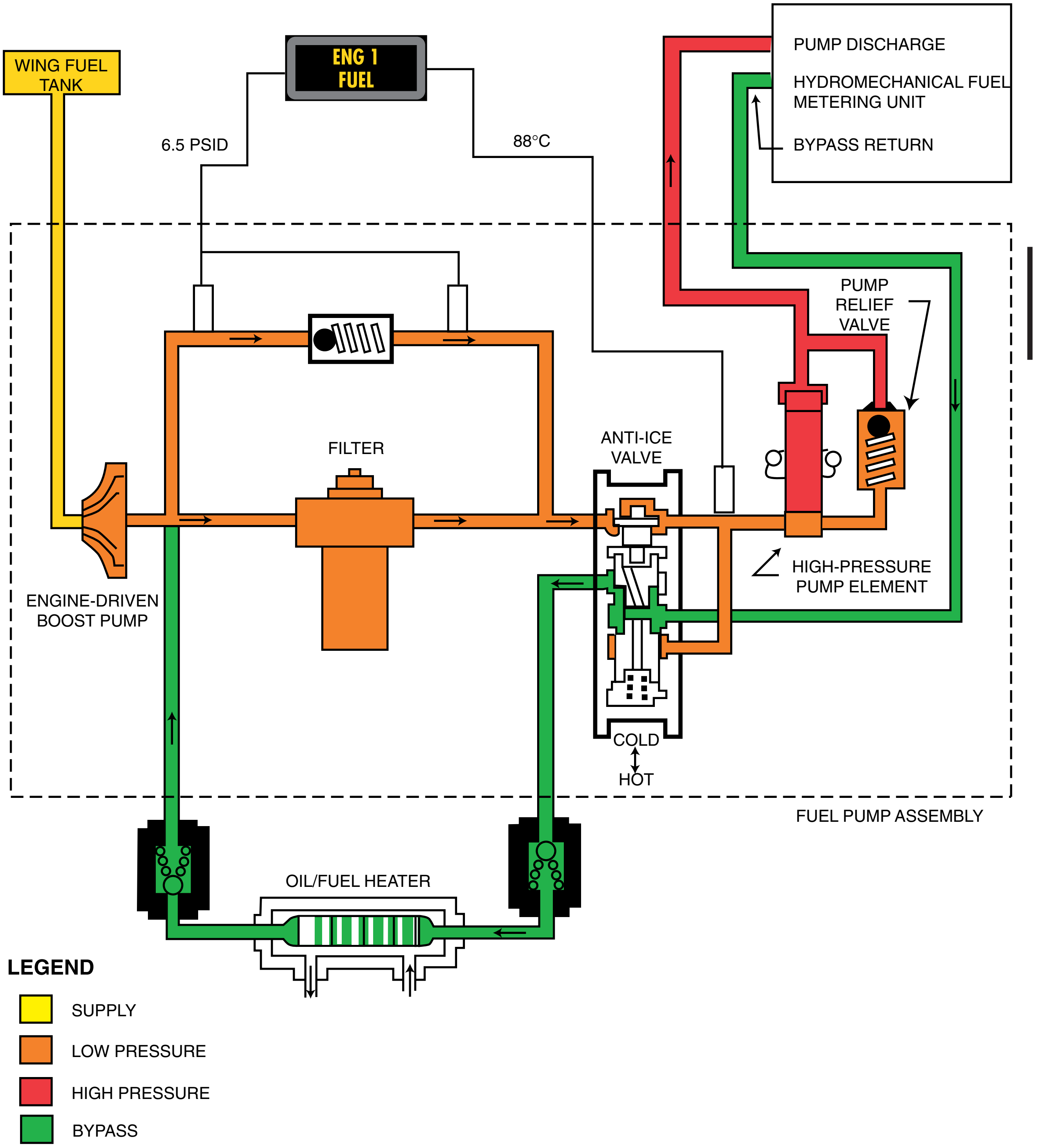


Figure 7-8. Fuel System

An operating fuel computer electrically controls fuel flow scheduling by setting the FCU metering section pressure drop according to throttle and engine inputs.

The FCU has two shutoff valves in series. The throttle actuates one valve and the electronic engine computer actuates the other valve. If the computer sense an ultimate overspeed condition, the computer closes the shutoff valve, fuel flow stops, and the engine shuts down.

Inlet Pressure and Temperature Sensor

An inlet pressure and temperature sensor is located on the cowling forward of the fan inlet. The sensor contains electrical elements for sensing inlet air temperature (Tt_2) and a pressure tap for sensing inlet air pressure (Pt_2). Both inlet parameters are required by the fuel computer. An electrical anti-icing element is contained in the sensor.

Digital Electronic Engine Control (Fuel Computer)

The -62 Digital Electronic Engine Control (DEEC) for the TFE 731-5BR engine is a closed loop control, that uses N_1 rather than N_2 as the control parameter. The -62 DEEC minimizes N_1 overshoots following engine acceleration and provides precise, repeatable N_1 control for all engine operating regimes. This control does not require engine trimming.

Two digital electronic engine controls (DEECs) are located in the rear equipment bay. Each DEEC controls the engine acceleration and deceleration. Separate ENG CMPTR switches on the overhead panel allow automatic (AUTO) or overspeed protection (OVSPD PROT) mode selection (Figure 7-9). Both DEECs receive 28 VDC from the PE busbar in the automatic mode (see Figure 7-10).

During acceleration and deceleration, the DEECs provide governing, limiting and scheduling response to the throttle and engine inputs. Inputs to each computer are:

- Engine Inlet Pressure (Pt_2)
- Engine Inlet Temperature (Tt_2)
- Interstage Turbine Temperature (ITT)
- N_2 Speed
- N_1 Speed
- Power Lever Angle (PLA).

Each DEEC provides appropriate output current to the torque motor of the associated FCU based on the various inputs.

Circuits within the DEEC monitor N_1 and N_2 continuously to provide overspeed protection. The computer commands the engine to shut down if the engine speed exceeds 107% N_1 or 109 % N_2. If an overspeed occurs, the primary overspeed circuit arms an electronic switch; the switch energizes the overspeed solenoid that cuts fuel to the engine. This function is called the ultimate "overspeed protection."

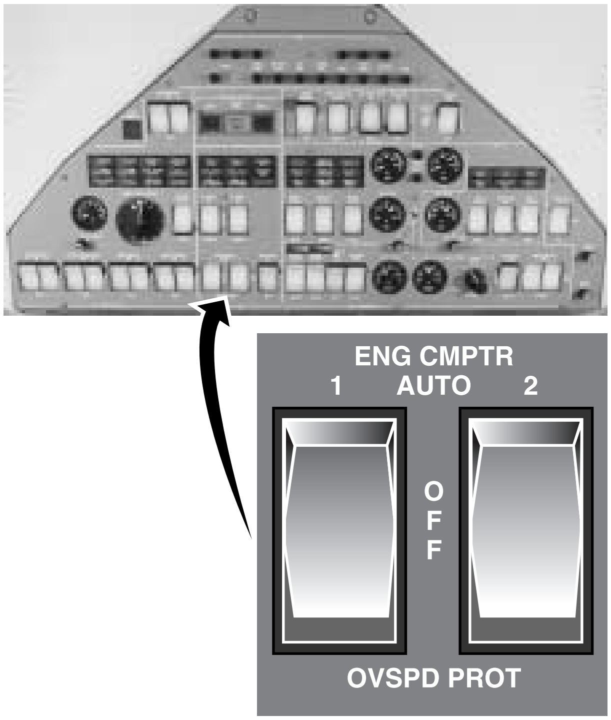


Figure 7-9. Engine Computer Switches

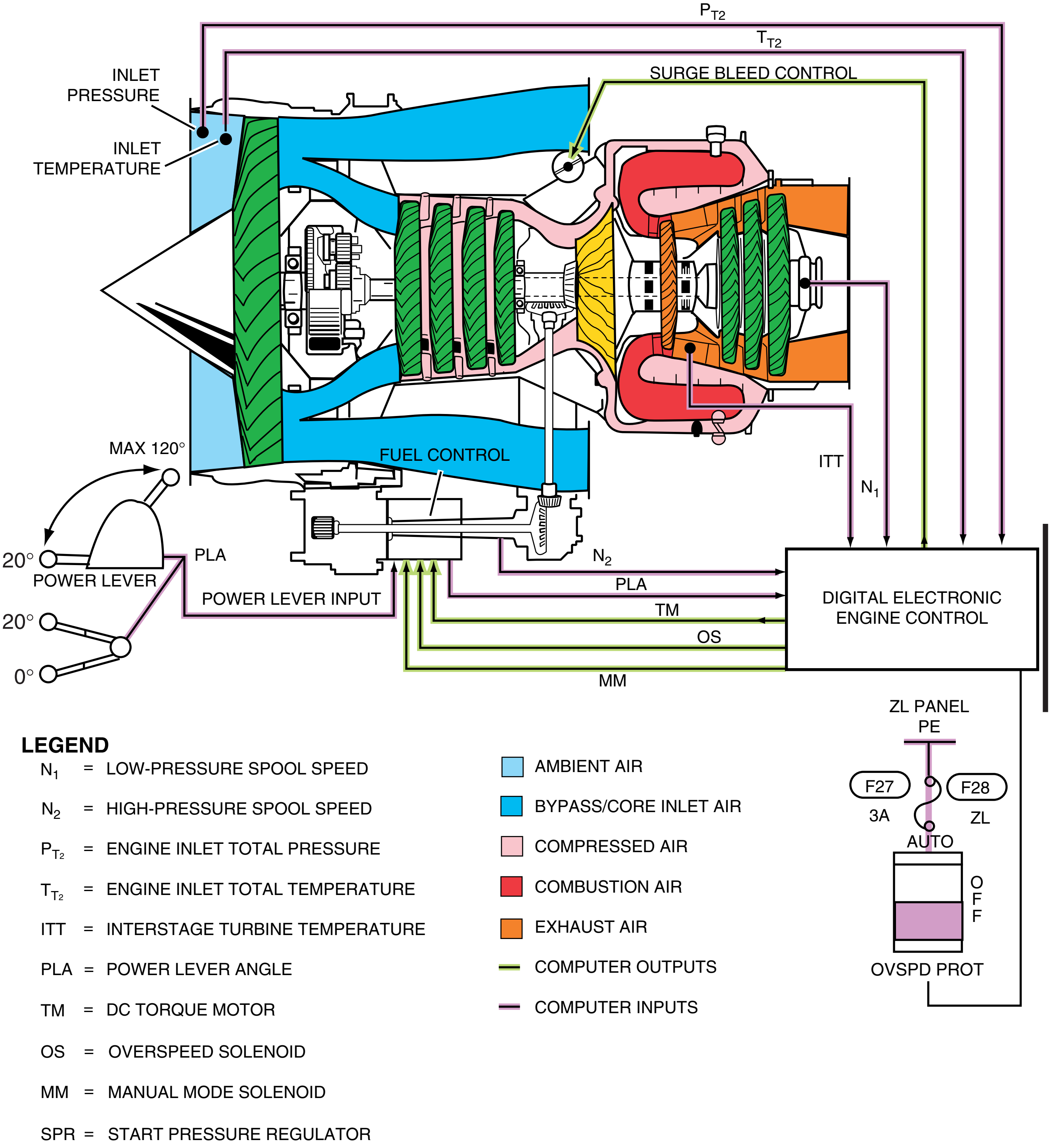


Figure 7-10. Fuel Computer—Engine Interface

If the electronic engine control malfunctions, the FCU on the engine automatically switches to the manual mode and the respective ENG CMPTR annunciator will become lit.

Surge Bleed Valve

Under certain conditions, gas turbine engines tend to surge and stall. For each compressor rpm, there is a relationship between the amount of air flow and the pressure gradient; a disturbance results in the engine surging. A surge bleed valve protects against this problem.

The DEEC controls the position of the surge bleed valve which is located between the LP compressor and the HP compressor, to prevent compressor stalls and surges. If the valve opens, compressed air flows into the bypass duct smoothing out the pressure gradient throughout the engine.

The DEEC normally positions the surge bleed valve fully open for start and idle conditions and fully closed for high rpm conditions. For transient rpm conditions, however, the DEEC modulates the surge bleed valve in response to impending stall conditions. With the DEEC off or failed, the surge bleed valve remains 1/3 open.

Fuel Flow Divider Assembly

The fuel flow divider is between the fuel control unit ant the fuel atomizers. During the engine start, the divider routes fuel at a reduced pressure to the primary atomizers. As the start sequence continues and the rpm increases, the fuel flow and pressure difference across the divider orifice increases; fuel passes into the secondary lines that supply the fuel atomizers.

Fuel Atomizers

Each engine uses twelve duplex (primary and secondary) fuel atomizers on two manifold assemblies; each manifold contains six duplex atomizers. Fuel swirls and breaks into microscopic droplets as it passes through the atomizer orifice into the combustion chamber. The primary and secondary fuel atomizers provide a finely atomized fuel spray pattern.

Fuel Heating

The fuel heater permits oil-to-fuel heat exchange to maintain the desired temperature and prevents ice formation in the fuel system from clogging the fuel pump assembly fuel filter. A portion of the engine fuel supply is diverted through the fuel heater by the thermostatically operated anti-ice valve located in the fuel pump.

An amber ENG FUEL 1 or 2 annunciator will become lit when the temperature of the fuel in the associated fuel pump becomes excessive or the fuel filter becomes clogged (see Figure 7-11).

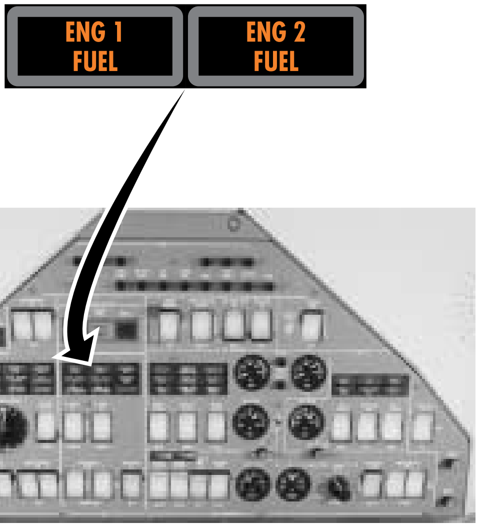


Figure 7-11. Engine Fuel Annunciators

Fuel Flow Indicating

The fuel flow indicating system for each engine consists of:

- Fuel Flow Transmitter
- Signal Conditioner
- Flow Rate Indicator
- Fuel Flow/Fuel Used Indicator.

The transmitter is a turbine-driven motor that rotates and generates an AC electrical signal as fuel flows past it. From the transmitter a signal conditioner converts the AC signal into DC for the indicators.

A fuel flow indicator with an integral fuel used indicator, on the center instrument panel displays fuel flow with a pointer moving over a graduated scale. The unit displays fuel flow in pounds-per-hour (PPH) from 0 to 2,200.

The fuel flow transmitter and indicators use 28 VDC power from the PS1 and PS2 busbars. The left system receives power from the PS1 busbar, and the right system receives power from the PS2 busbar.

POWER CONTROL

INTRODUCTION

The power control system on the engine consists of (see Figure 7-12):

- Throttle Levers
- High Pressure (HP) Fuel Cocks
- Fuel Control Unit
- Electronic Engine Computer
- Engine Synchronization.

Throttles

Each throttle lever on the center pedestal mechanically connects through cables and a teleflex control to a fuel control unit.

Movement of the throttle directly drives the fuel control unit from idle to full power. In response to throttle movements and engine parameters, the digital electronic engine coontrol (DEEC) provides an electric signal to the hydromechanical fuel control unit torque motor. The fuel control unit either decreases or increases the flow of fuel to the engine to provide overspeed and overtemperature protection.

With the DEEC failed and OVSPD PROT selected, or OFF selected, through the ENG CMPTR switch on the overhead panel, the throttle lever directly controls the engine power through the fuel control governor.

The HP cock positions are in relationship to the angle of rotation of the control shaft on the FCU. The full aft (0°) position is the engine fuel cutoff position. The idle (or engine start) position is forward at 20°. To move the throttle valve from idle to cutoff or from cutoff to idle, the HP fuel cock lever must be lifted. The throttle has unrestricted travel from idle to full thrust, when the HP fuel cock lever is in the forward position.

Audible Warnings and Interlocks

The throttle levers operate two microswitches through a cam on the throttle cable drum shaft. Reducing power below 65 % N_1 rpm with the landing gear not locked down below 150 kts completes a circuit that sounds a warning horn.

Increasing power with the air brakes extended while the landing gear is down will complete a circuit that sounds a warning horn.

A mechanical locking device interconnects both throttle levers to prevent simultaneous application of engine power above 60 % N_1 with the elevator gust lock installed. Although one throttle at a time can be advanced to any setting.

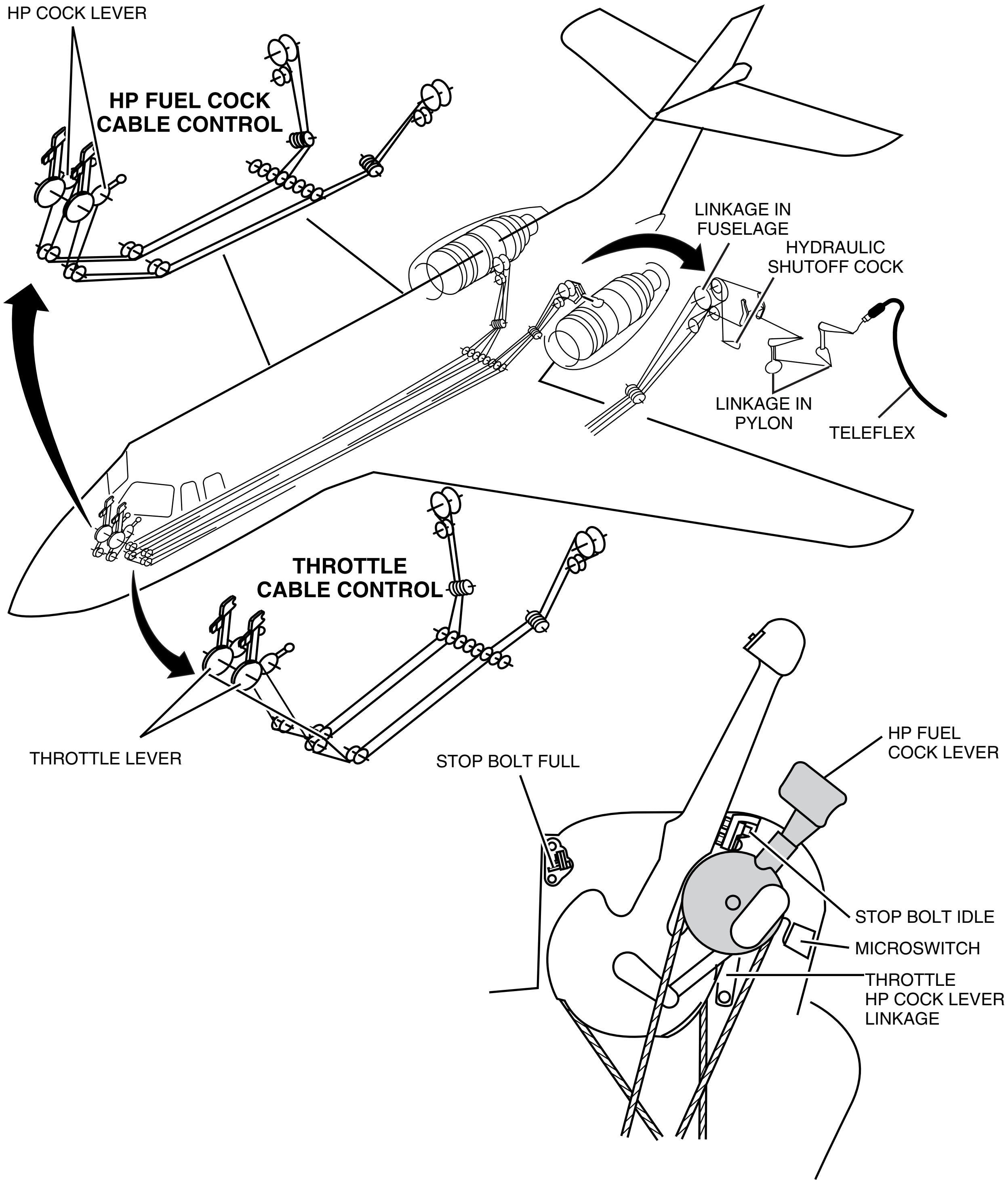


Figure 7-12. Fuel Controls

HP Fuel Cocks

Each high pressure (HP) fuel cock control lever connects mechanically through cables and teleflex controls to the fuel control unit. The levers control the opening and closing of the fuel control units from off (no fuel flow) to the idle throttle setting. The levers also connect mechanically with the hydraulic supply valves. Closing of a HP fuel cock simultaneously isolates the respective engine's hydraulic fluid supply.

A cam and spring at the OFF (0°) and ON (20°) positions of each HP fuel cock control lever mechanically lock the levers in either position. Before moving the lever, the knob must be pulled out to unlock it. The lever automatically locks once it reaches the OPEN or CLOSED positions.

Microswitches within each lever control power to the engine igniter units. In OFF, a circuit opens to remove power to the igniter unit. In ON, the switch closes to supply power to the respective unit.

A red warning light is above the ON position of each HP fuel cock control lever. The light will become lit in combination with the fire warning system as a reminder to close the respective cock.

Engine Synchronizer

The engine synchronizer provides synchronization of the engines. Using the three position ENG SYNC switch (Figure 7-13), synchronization of either the low pressure fan (N_1) or the high pressure turbine (N_2) speeds can be selected in the cockpit. The left engine (No. 1) is the master engine and the right engine (No. 2) is the slave.

The system compares either the N_1 or N_2 speeds of the engines. The synchronizer processes speed signals from each engine and provide a trim signal to the electronic engine computer of the slave engine to reduce an speed difference.

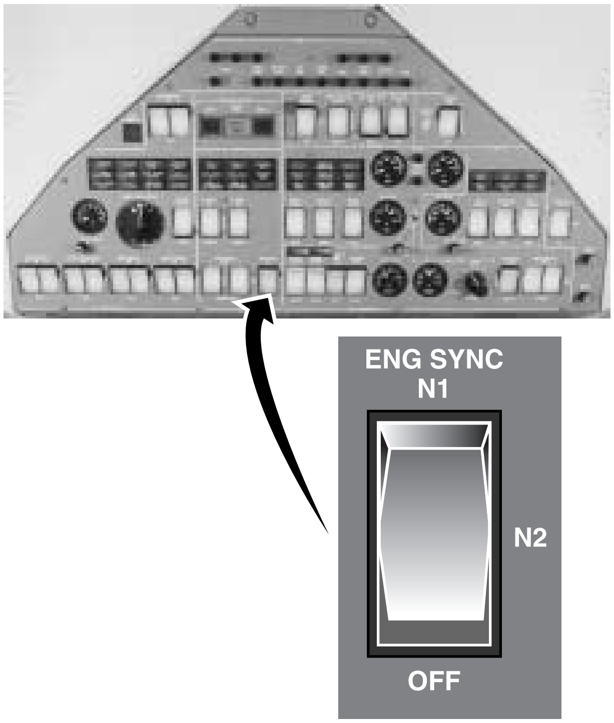


Figure 7-13. Engine Synchronizer Switch

Synchronization has limited authority and can occur only when speed differential is within the authority range. The maximum authority range is ±2.5 % N_2 at throttle midrange; authority range decreases as engine speed increases or decreases from the throttle midrange. The synchronizer has no effect at the full throttle settings.

The OFF position of the switch removes the DC power from the synchronizer; the N_1 and N_2 positions select the spool that will be used for synchronization. Operation of the synchronizer requires both engine computer switches to be in the AUTO position.

With APR armed the synchronizer is inoperative. When a synchronizer is switched off, the N_1 rpm indicator shows N_1 rpm compensated for the thrust of the engine. When the synchronizer is switched to N_1 or N_2 the N_1 rpm indicator shows the N_1 rpm.

The engine synchronizer system uses 28 VDC from the PE busbar.

ENGINE BLEED AIR AND VENTILATION

INTRODUCTION

Air is bled from two stages of the engine compressor to provide supplies for (see Figure 7-14):

- Nacelle Inlet Cowl Anti-icing
- Aircraft Services.

Ram air is used to ventilate the area of the cowling surrounding the engine compressor stages between the front and rear firewalls.

BLEED AIR

Anti-icing

An ENG ANTICE ON-OFF switch (see Figure 7-15), located on the roof panel ice protection section, is provided for each engine. With either or both switches selected to ON, an ICE PROT SELECTED annunciator on the MWS panel is lit.

Each switch controls a servo-operated anti-icing on-off valve. When ON is selected, the following events occur:

- The anti-icing valve deenergizes open and high pressure air from the HP compressor and is ducted forward to anti-ice the nacelle inlet cowl.
- Electrical power is supplied to PT2 and TT2 sensor probe heaters.
- In flight, the engine digital computers are reset to a schedule that incorporates a raised idle rpm to compensate for the effect on thrust.
- The temperature provided by the A panel windscreen film temperature is uplifted from the normal setting to ensure adequate anti-icing performance.

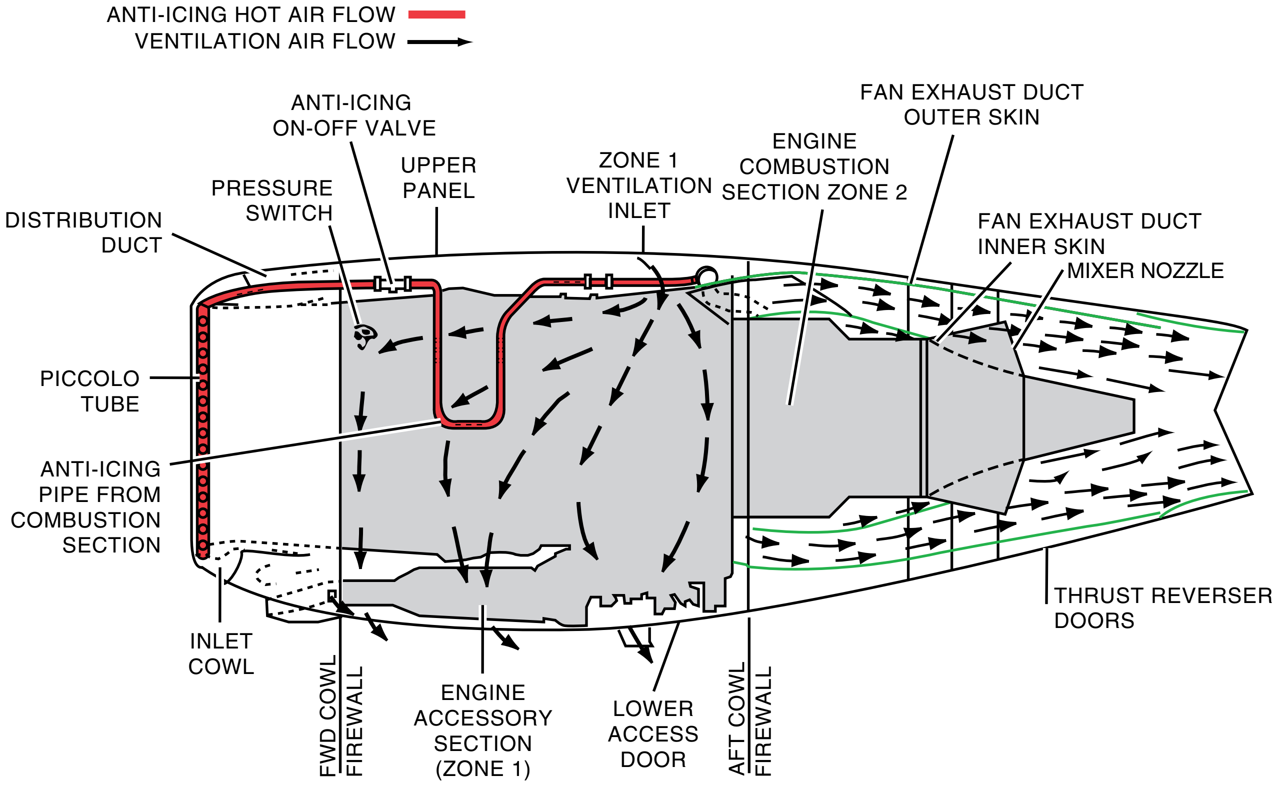


Figure 7-14. Engine Air Bleeds and Ventilation

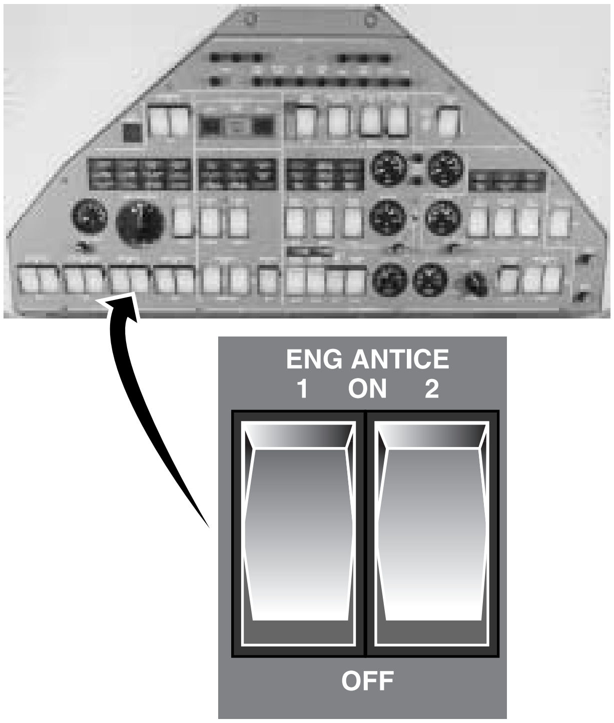


Figure 7-15. Engine Anti-ice Switch

A pressure switch, which operates at 6 psi, is tapped into the air bleed pipe from the engine. When the air pressure is low, the switch operates to cause an ENG 1 (or 2) A/ICE annunciator to be lit on the MWS panel.

The anti-icing valve is energized to the closed position (with busbar energized and ENG ANTICE switch selected to OFF), and is biased to the open position. This provides anti-icing fail-safe operation in the event of an electrical malfunction.

Prior to the opening of an anti-icing valve, or during any subsequent system failure, the pressure switch will register a low pressure condition and the appropriate ENG A/ICE annunciator will be lit at the MWS dim pre-set level (Figure 7-16). The annunciator will remain lit at the dim level after the switch has been selected to ON, until the nacelle anti-icing air supply rises above 6 psi and the pressure switch contacts change over. Then the annunciator goes out.

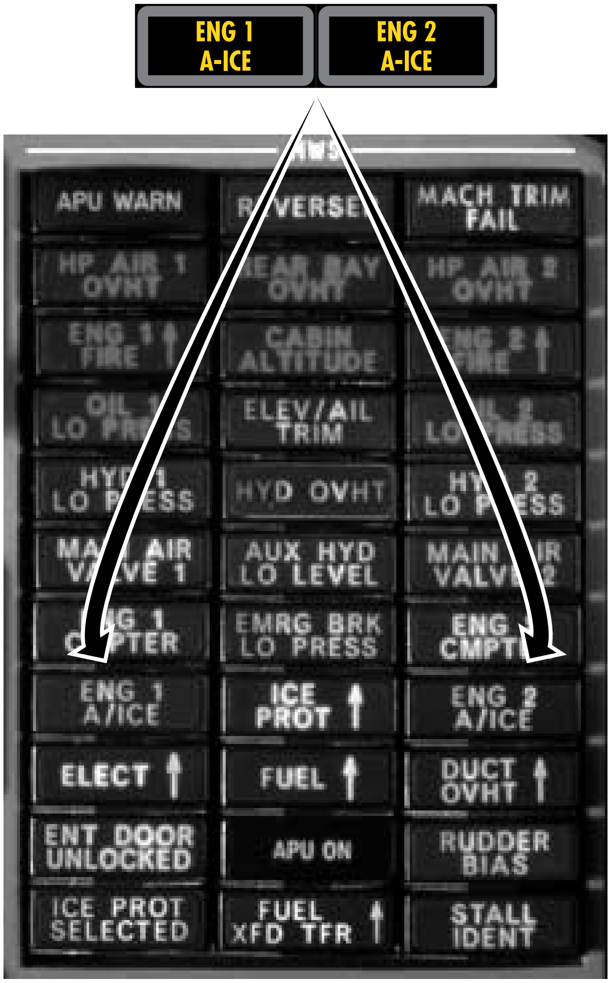


Figure 7-16. Engine Anti-ice Annunciator

However, a timer in the circuit ensures that the annunciator will brighten to full intensity, should the pressure switch not operate within two seconds. The time delay also inhibits ‘nuisance’ flashing of the annunciator during normal system operation.

Electrical Power Supplies

DC supplies for the engine anti-icing system are taken from busbar PE.

Aircraft Services

Each engine has two bleed ports used for aircraft services:

- One low pressure (LP) bleed port on the inboard side of each engine takes bleed air aft of the last stage of the axial compressor.
- A high pressure (HP) bleed port takes air from downstream of the centrifugal compressor.

Each bleed air system contains check valves that prevent air from returning to the engines at low engine speeds or during engine shutdown.

The aircraft uses LP bleed air for air conditioning, normal pressurization, emergency pressurization and hydraulic reservoir pressurization.

HP bleed air supplements LP bleed air supplying these services at low power settings. The rudder bias system receives power from the LP bleed air system only.

AUTOMATIC PERFORMANCE RESERVE (APR)

INTRODUCTION

The APR system, when armed, allows an automatic increase in engine performance in the event of an engine failure or transient fault during takeoff, or during a "go-around" following a single-engined approach.

There will be no increase in thrust if the ambient conditions are such that the engines are N1 rpm limited (at normal rating).

The APR system is powered from the PE busbar and comprise, two mechanically latching, "push on/push off," control switches in the flight compartment plus an APR/Synchronizer control unit in the rear equipment bay.

CONTROLS

Two control switches, the LH one labeled APR O/RIDE and the RH one APR ARM are mounted on the pilot's center instrument panel (see Figure 7-17).

The ARM switch has an APR ARM white annunciator which will be lit when the switch is pressed to arm the system and will not be lit when the switch is pressed again to disarm the system.

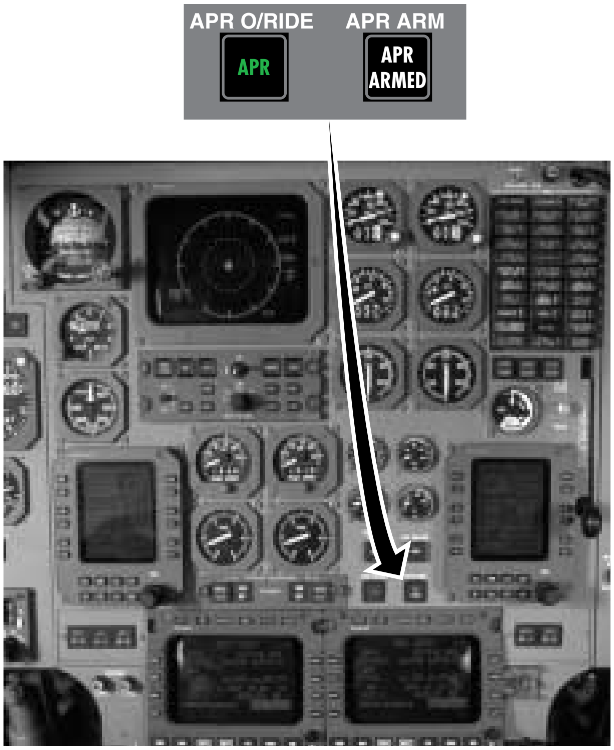


Figure 7-17. APR Controls

The O/RIDE switch has an APR green annunciator which will become lit when the system is armed and triggered, either automatically by N_2 difference or manually by pressing and latching the O/RIDE switch.

OPERATION

See Figures 7-18 through 7-20.

When armed, APR is triggered automatically if the engine N_2 signals (received from the synchronizer and compared by the APR controller), differ by more than 5% N_2.

Increase in thrust is obtained by allowing the engines to be run at higher operating limits; the N_2 rotor speed increasing approximately 0.75% (max) and the interstage turbine temperature (ITT) rising approximately 18° C (max).

Automatic Mode

In auto mode the system functions in conjunction with the engine synchronizer system.

When armed, APR will be triggered automatically if the engine N_2 speed signals received from the synchronizer and compared by the APR control unit differ by more than 5% N_2 (approximately 12% N_1).

Both engine fuel computers limit maximum takeoff power to the APR rating although

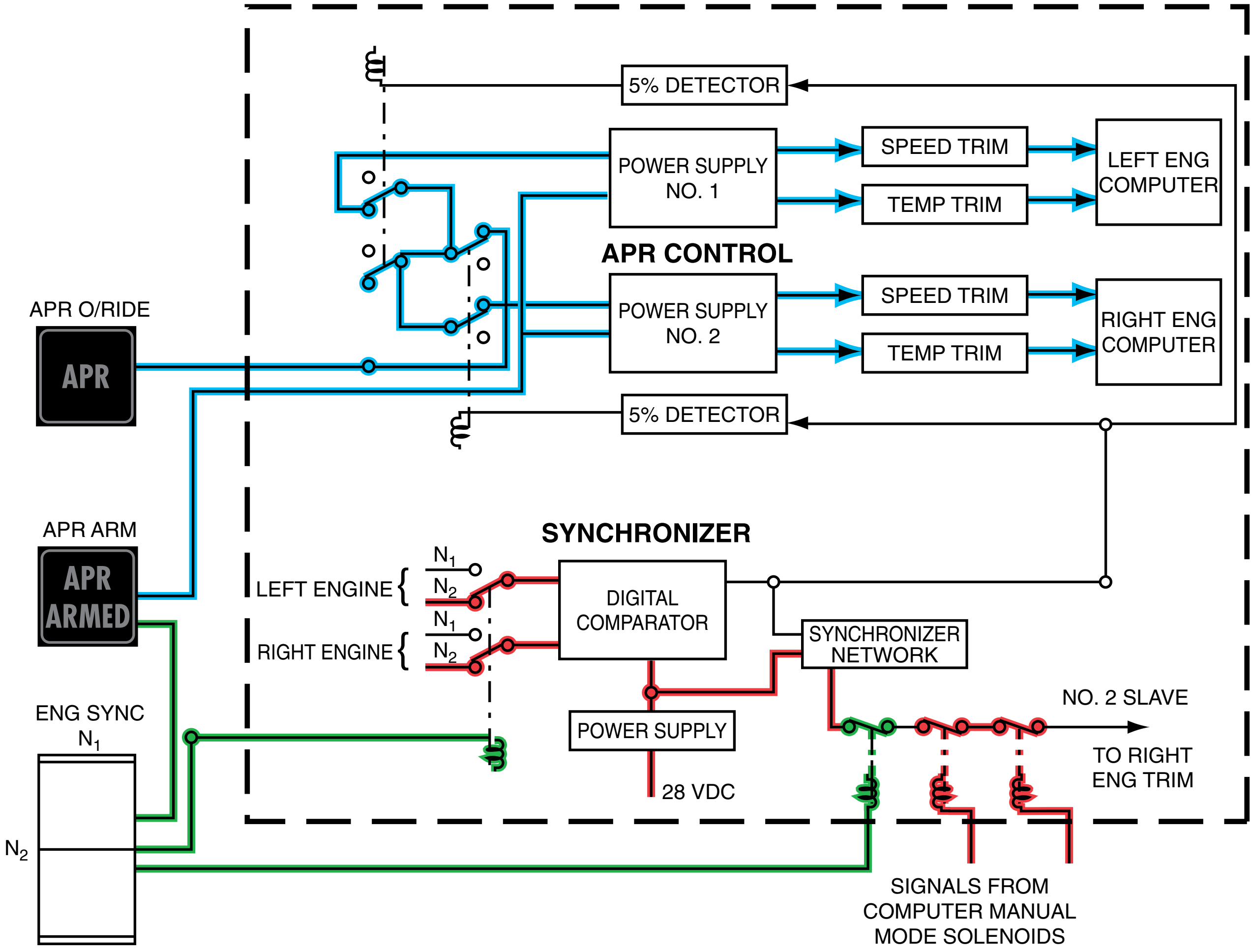


Figure 7-18. APR—OFF SYNC—N_2

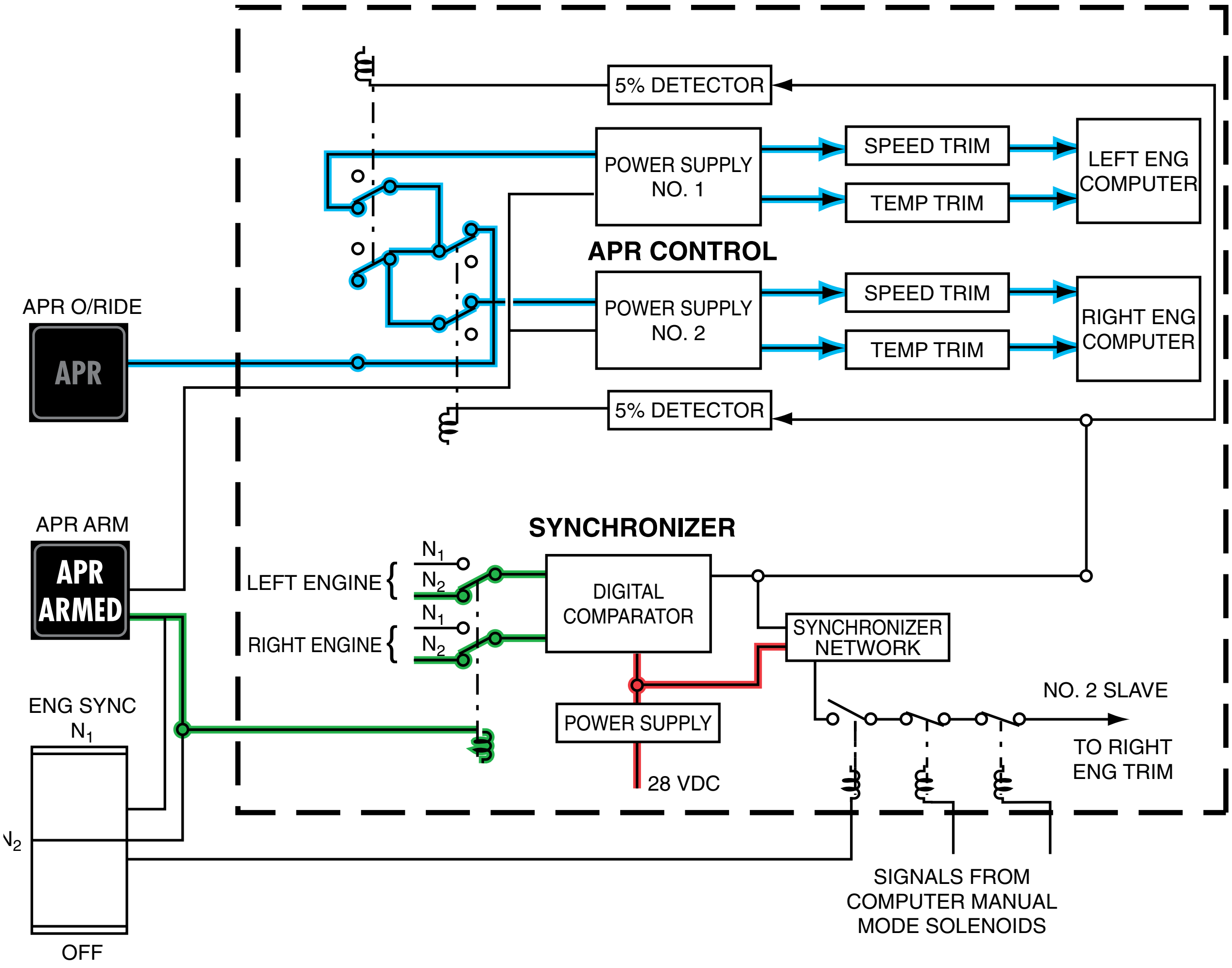


Figure 7-19. APR—Armed

under normal conditions, the APR control unit electrically trims down the settings to the lower normal ratings. When the APR is triggered the control unit allows the computers to reset to the APR ratings; normal engine response to the throttle movement is retained.

Override Mode

When armed, to cater for an auto mode malfunction, the system may also be triggered by manual selection of APR override (O/RIDE).

NOTE

There is a penalty in terms of engine life whenever an engine operates at APR levels—see the *Aircraft Maintenance Schedule* for further details. Therefore override selection of APR should only be made when it is essential to do so.

To cancel APR mode after override selection, both the APR O/RIDE and APR ARM switches should be delatched. If only the APR ARM switch is delatched, APR will be cancelled, but will be triggered again as soon as the APR ARM switch is pressed again.

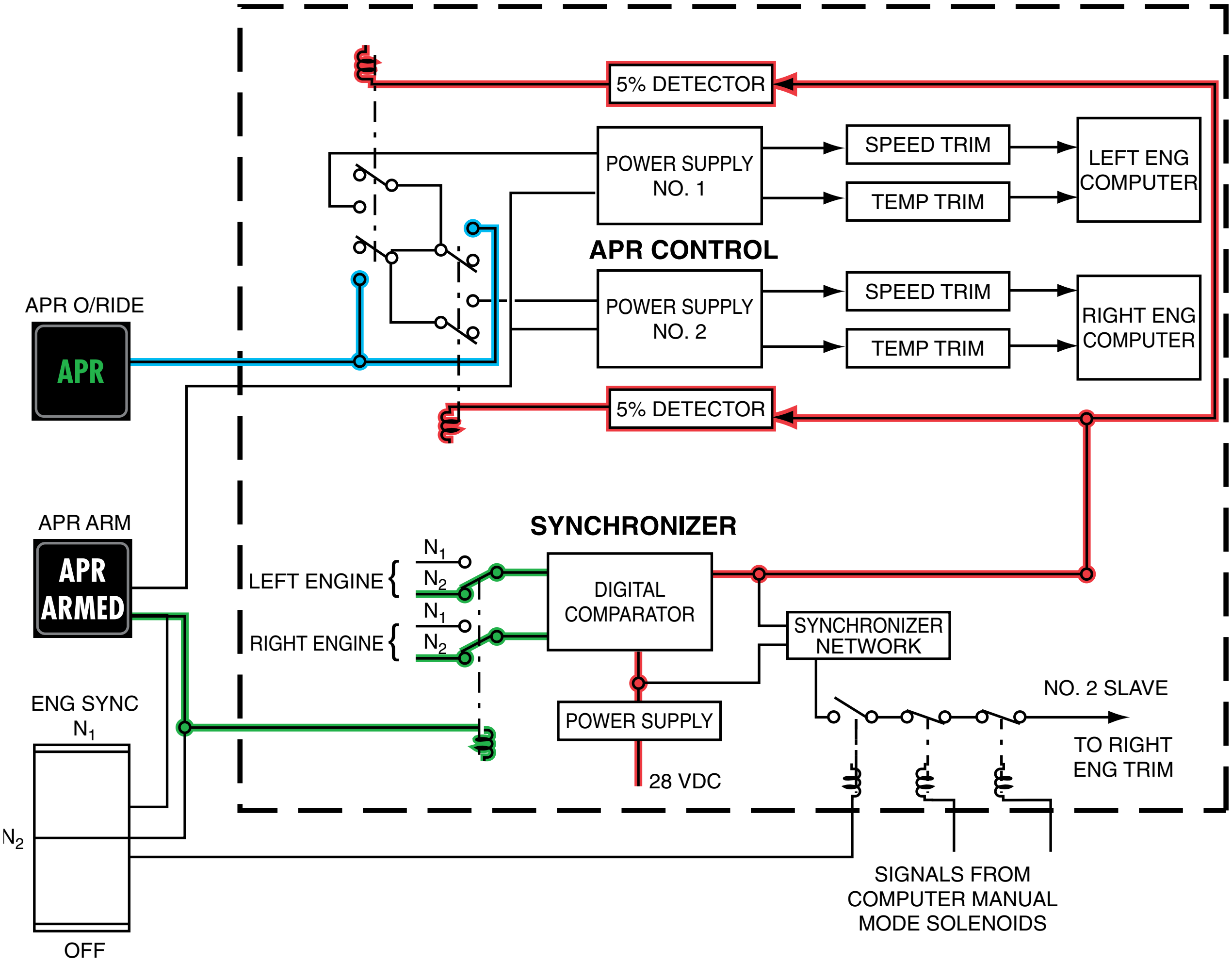


Figure 7-20. APR—Armed 5% N_2 Sensed

APR/Synchronizer Controller

Both engine fuel computers limit maximum takeoff power to the APR rating. Under normal takeoff conditions the APR/synchronizer controller electrically trims down the power settings to the normal (lower) rating.

When APR is triggered the controller allows the computers to reset the APR (higher) rating; the engine response to throttle movement being retained.

Failure modes

Failure of electrical power to the APR/synchronizer controller will also result in both fuel computers resetting to the APR ratings.

The synchronizer must be serviceable to supply N_2 comparisons for the APR auto mode. Manual O/RIDE is independent of N_2 signals.

An engine fuel computer must be serviceable and selected to AUTO in order to respond to APR. Should an engine computer fail, or is selected to manual, with APR armed, APR will be triggered in the other engine if engine speed differences exceed 5% N_2.

ELECTRICAL POWER SOURCES

PE bus bar

- APR control unit
- Computer-AUTO, 1 and 2
- Computer-OVSPD PROT, 1 and 2
- ENG FUEL warning, 1 and 2
- HP cocks warning via fire detection system
- Engine speed synchronizer
- Oil pressure indicators 1 and 2
- Oil temperature indicators 1 and 2
- OIL LO PRESS annunciators 1 and 2

PS1 bus bar

- No. 1 N1 RPM indicator*
- No. 1 N2 RPM indicator*
- No. 1 ITT indicators*
- No. 1 fuel flow indicator

*PE during internal engine start

PS2 bus bar

- No. 2 N_1 RPM indicator*
- No. 2 N_2 RPM indicator*
- No. 2 ITT indicator*
- No. 2 fuel flow indicator

*PE during internal engine start

THRUST REVERSER SYSTEM

GENERAL

The Hawker-800 XP airplanes are equipped with a Dee Howard TR5000BA Thrust Reverser System. The system includes a thrust reverser assembly at the rear of each engine nacelle and various controls and indicators located within the cockpit.

THRUST REVERSER ASSEMBLY

Each thrust reverser assembly (Figure 7-21) is an electromechanically controlled, hydraulically operated, target-type thrust reverser consisting of:

- Upper and lower clamshell doors located on the rear of each engine nacelle
- Hydraulic actuators and associated linkages that position the reverser doors upon either stow or deploy commands
- Latches which hold the reverser doors in the stowed position until released by sequentially actuated switches and relays
- Fan flaps within each fan duct, which, in reverser mode, aid in temperature reduction on the doors. They deploy and stow in conjunction with the doors.
- Thrust reverser accumulator (1,500 psi minimum).

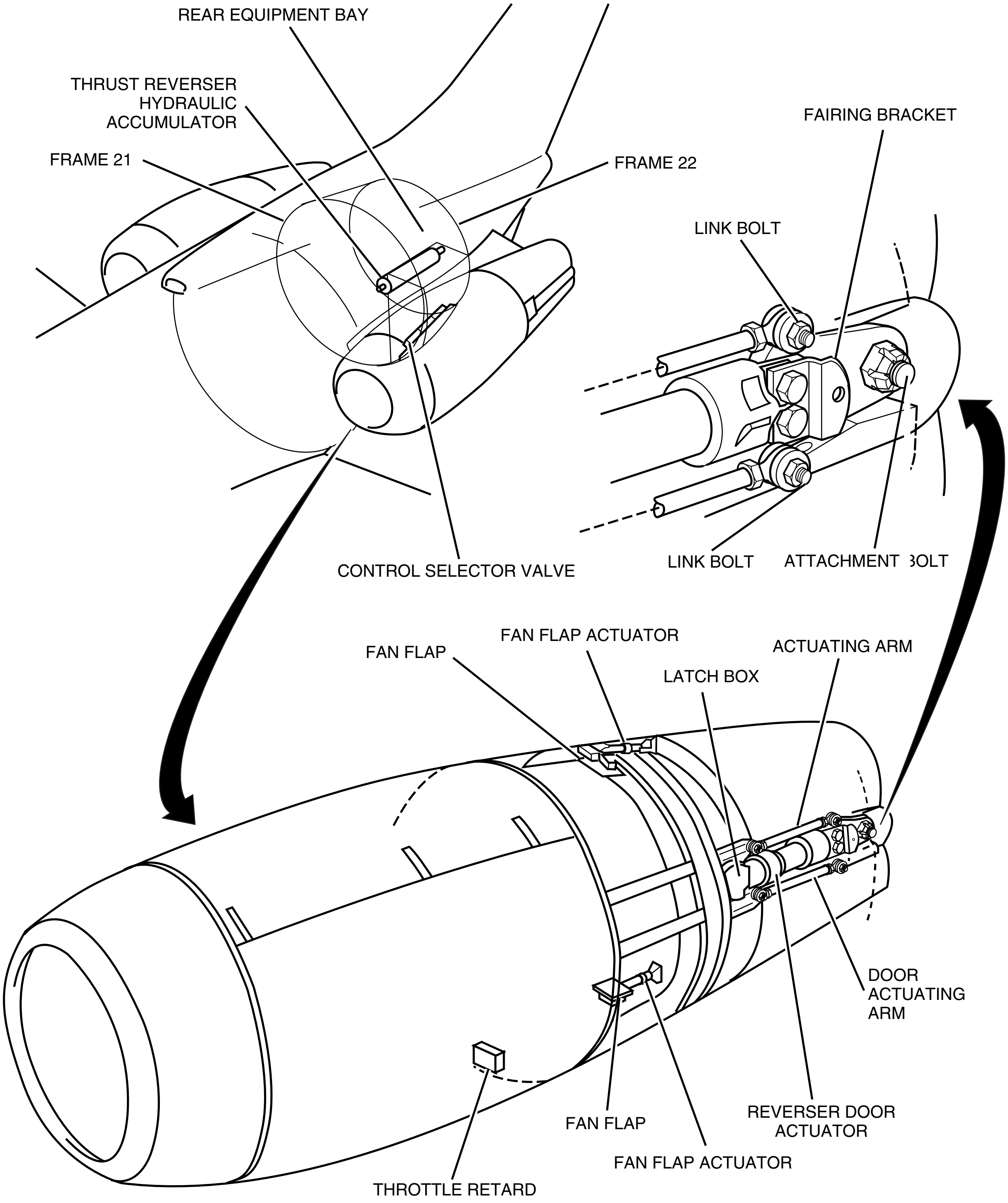


Figure 7-21. Thrust Reverser Assembly Components

CONTROLS AND INDICATORS

An electrical control panel, located on the lower section of the center instrument panel (Figure 7-22), contains L/H and R/H THRUST REVERSER POWER switchlights which illuminate ARM or OFF. Amber UNLCK and green REVRS lights are also included for each assembly and are found on this panel.

Thrust reverser levers are mounted piggyback on the normal throttles (Figure 7-23). Each lever provides control for stow, deploy, and the modulation of reverse thrust from reverse idle to maximum reverse thrust for its respective engine. Each throttle has a solenoid-operated throttle balk (lockout) which mechanically prevents selection of reverse thrust until the reversers are fully deployed. There is a stop on each throttle to mechanically limit the obtainable reverse thrust when the thrust reverser lever is increased to its maximum thrust position. An automatic throttle retard system is incorporated in the event of a stow or deploy malfunction.

An amber REVERSER annunciator, located on the top center column of the MWS panel (Figure 7-22), indicates the existence of a malfunction of either reverser.

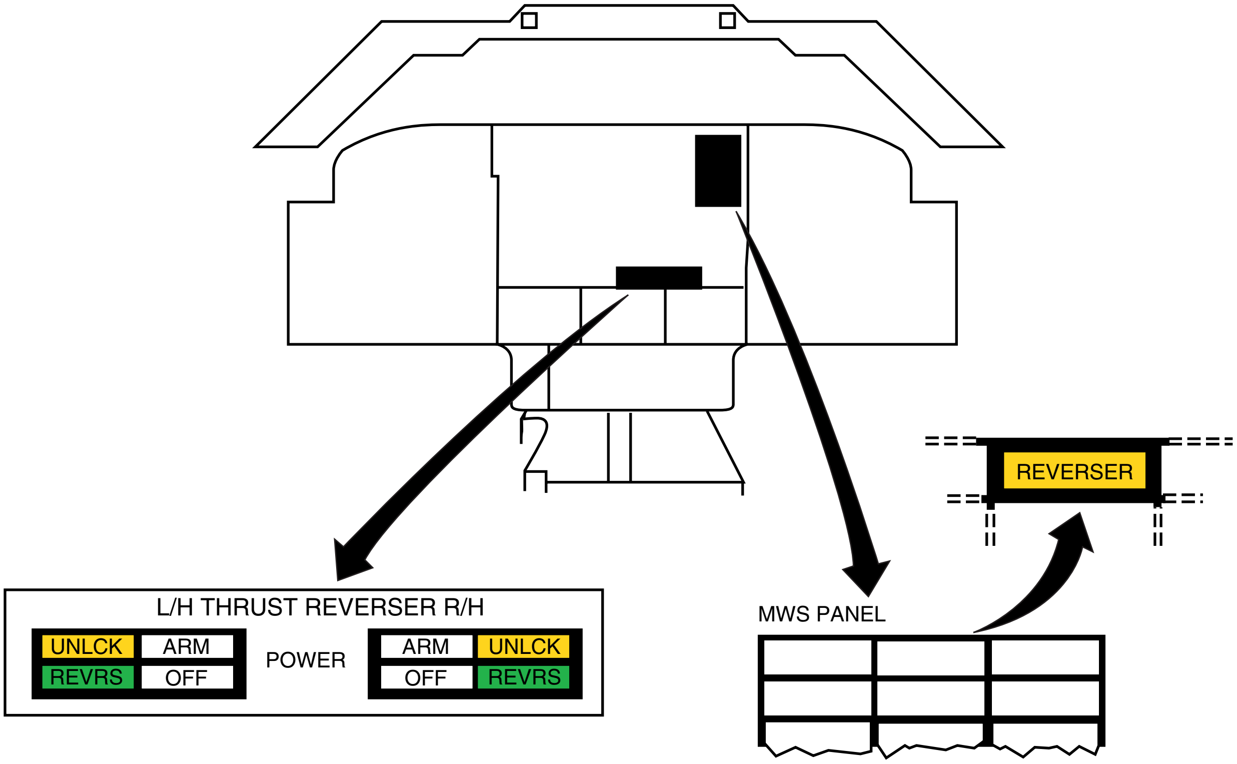
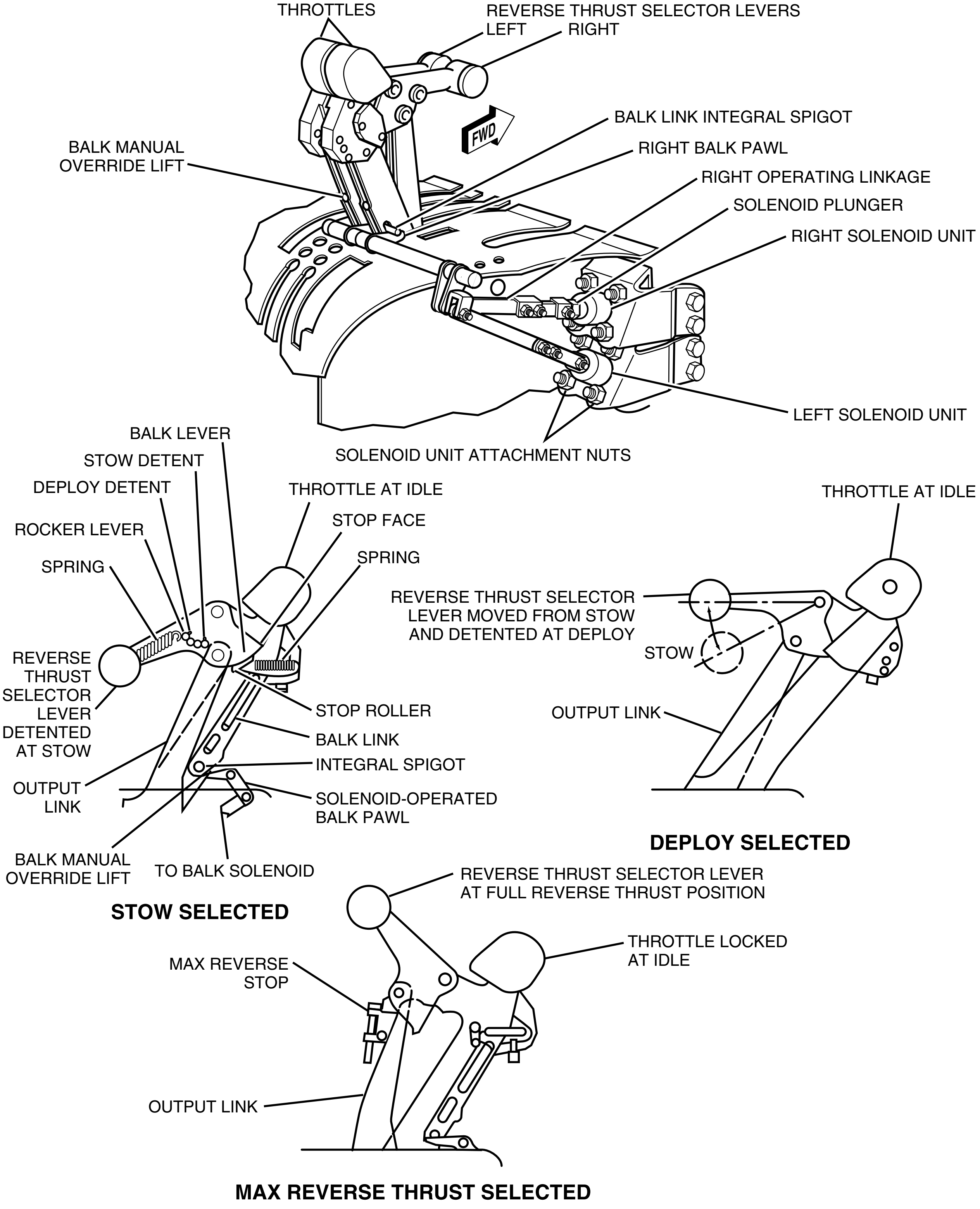


Figure 7-22. Thrust Reverser Control Panel and MWS Panel



STOW SELECTED

DEPLOY SELECTED

MAX REVERSE THRUST SELECTED

Figure 7-23. Thrust Reverser Throttle Controls

SYSTEM OPERATION

General

There are three phases of reverser operation:

- Initiation
- Deployment
- Stowing

Figure 7-24 shows the thrust reverser schematic in the deenergized condition.

Initiation

When the POWER switch is depressed (Figure 7-25) the ARM light will be illuminated, provided that the following conditions are met:

- Weight-on-wheels switches recognize an on-the-ground condition.
- Throttle lever is in the IDLE position
- Hydraulic pressure is available (minimum of 400 psi).

Deployment

The reverser can now be deployed when the thrust reverser lever is moved to the deploy position. The latch and stow solenoids are energized to provide pressure to the latch actuators and the primary actuators to provide an unlatched and overstow condition. Pressure is also directed to the throttle retard actuator (Figure 7-26). The unlocked annunciator will now be illuminated.

NOTE

The tendency of the exhaust forces on the doors is toward deploy whenever the engine is running. When engaged, the latches cannot be disengaged unless the overstow condition can be realized.

When the unlatch switches operate, the stow solenoid is deenergized and the deploy solenoid is energized. This retracts the primary actuators, which deploy the reverser doors and extend the fan flaps into the engine bypass airstream. The UNLCK and REVRS annunciators are now illuminated (Figure 7-27).

After initial deployment, the reverse thrust lever balk is released and additional reverse thrust may be commanded by pulling the levers toward maximum reverse. A mechanical stop is set at a predetermined thrust setting.

A deploy command inhibits operation of rudder bias to prevent rudder movement due to asymmetric thrust. The airbrake warning horn and the ELEV/AIL trim annunciator is also inhibited.

Stowing

Moving the reverse thrust lever toward stow deenergizes the stow/deploy relay and closes the unlatch solenoid (Figure 7-28). Deploy power is removed from the latch switches, the stow valve solenoid is energized, and hydraulic power is routed to the close side of the primary actuator, closing the reverser doors. Fan flaps are spring-loaded closed, and the throttle retard actuator is limited to IDLE. When the reverser doors reach fully locked, stow pressure is removed.

Autostow

If two latch position switches on the same side (inboard or outboard) of one reverser indicate an unlatched condition, a 28-VDC signal energizes the isolation and stow valves to automatically provide stow pressure, regardless of the position of the POWER switch. The ARM, UNLCK, and REVERSER annunciators automatically illuminate, and the affected throttle is moved toward IDLE. This autostow procedure continues until one of the two latches senses a latched signal (Figure 7-29).

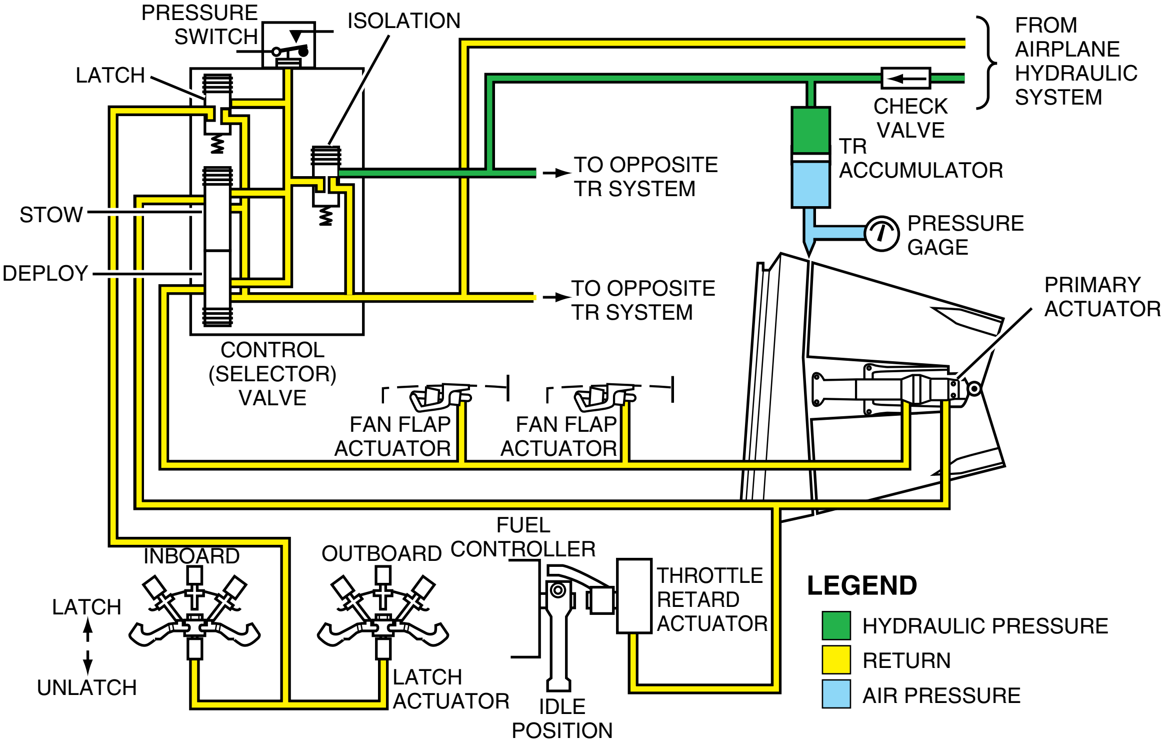


Figure 7-24. Thrust Reverser System—Deenergized

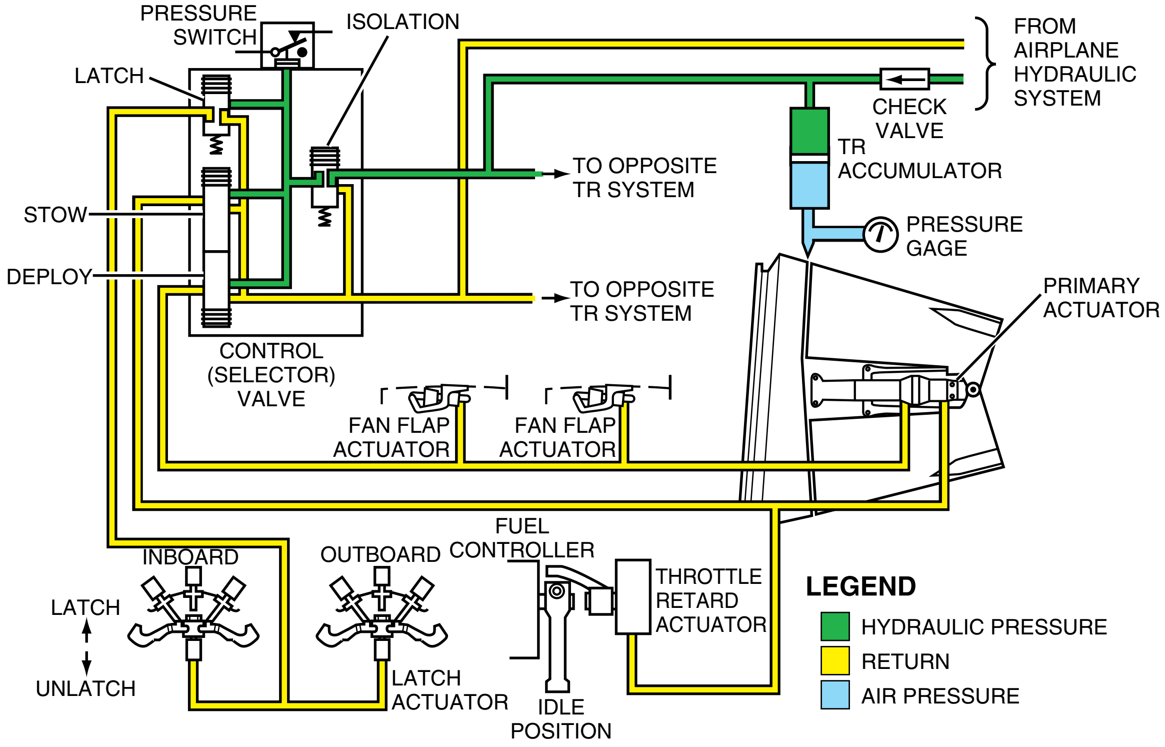


Figure 7-25. Thrust Reverser System—Armed

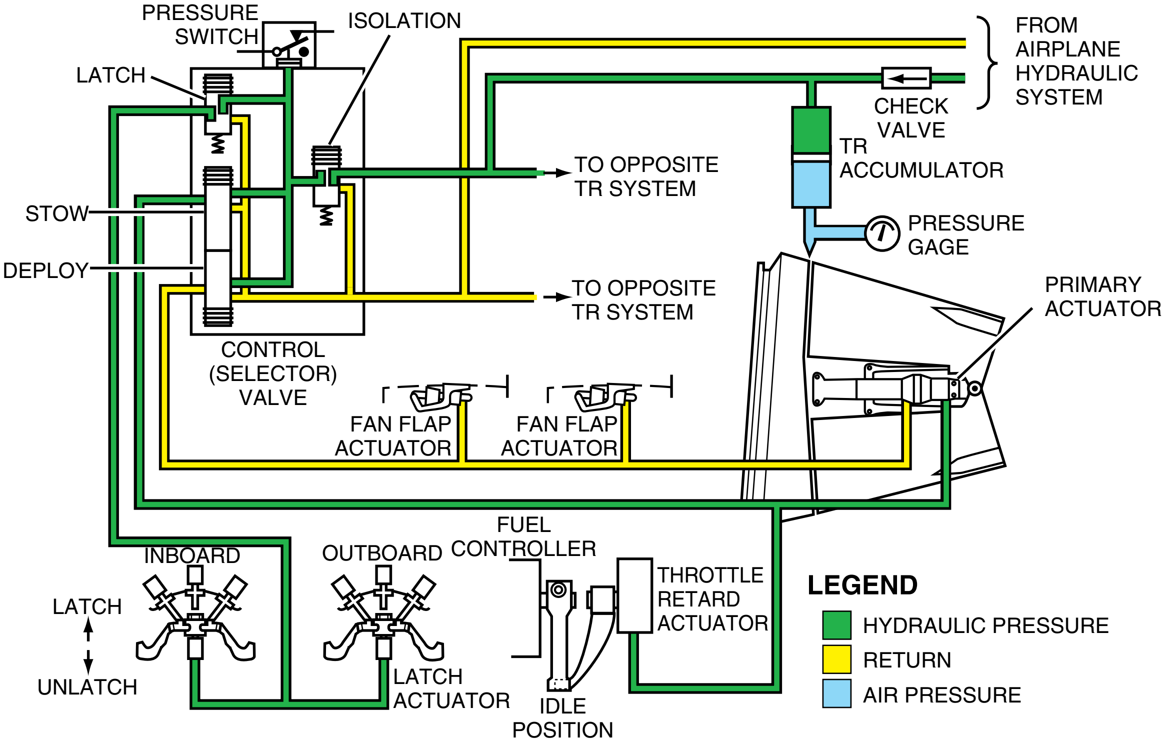


Figure 7-26. Thrust Reverser System—Overstow and Unlatch

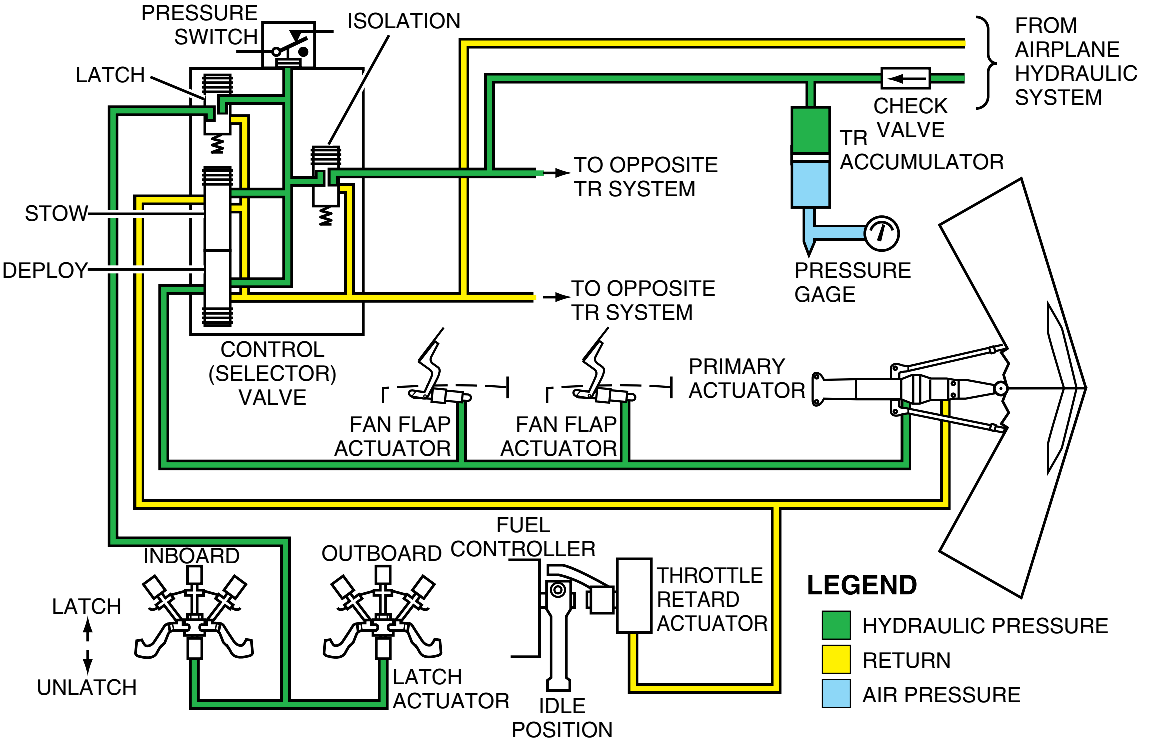


Figure 7-27. Thrust Reverser System—Deploy

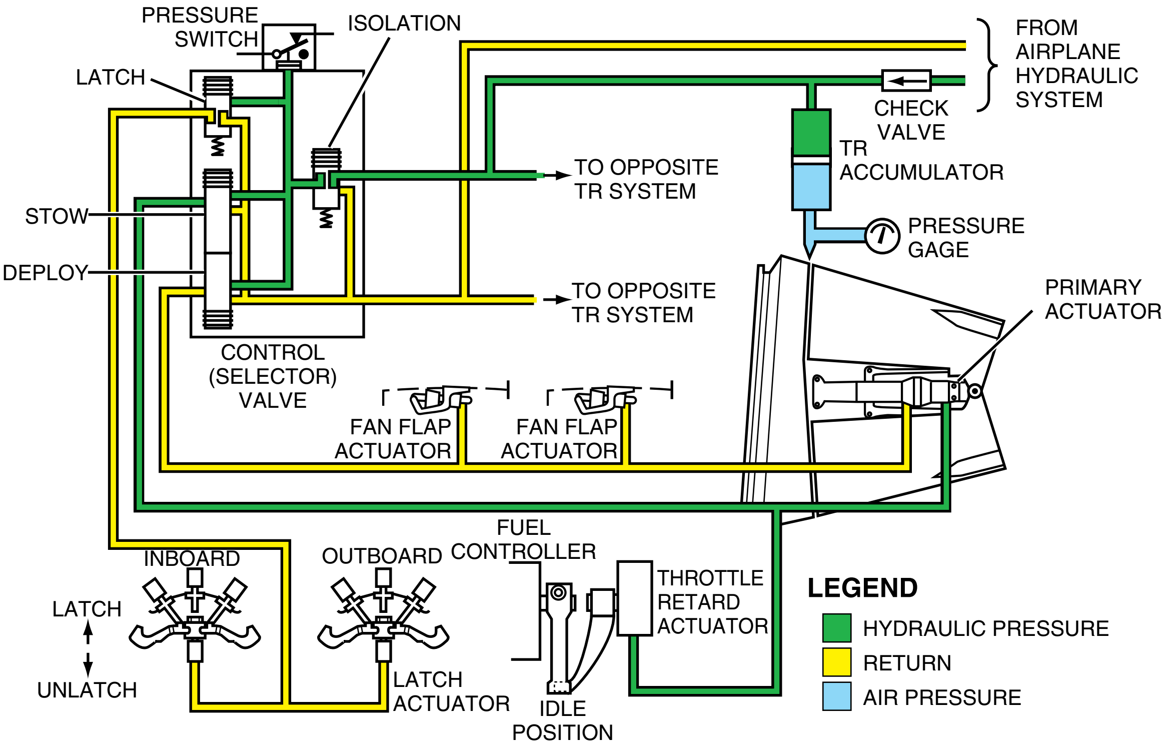


Figure 7-28. Thrust Reverser System—Normal Stow Cycle

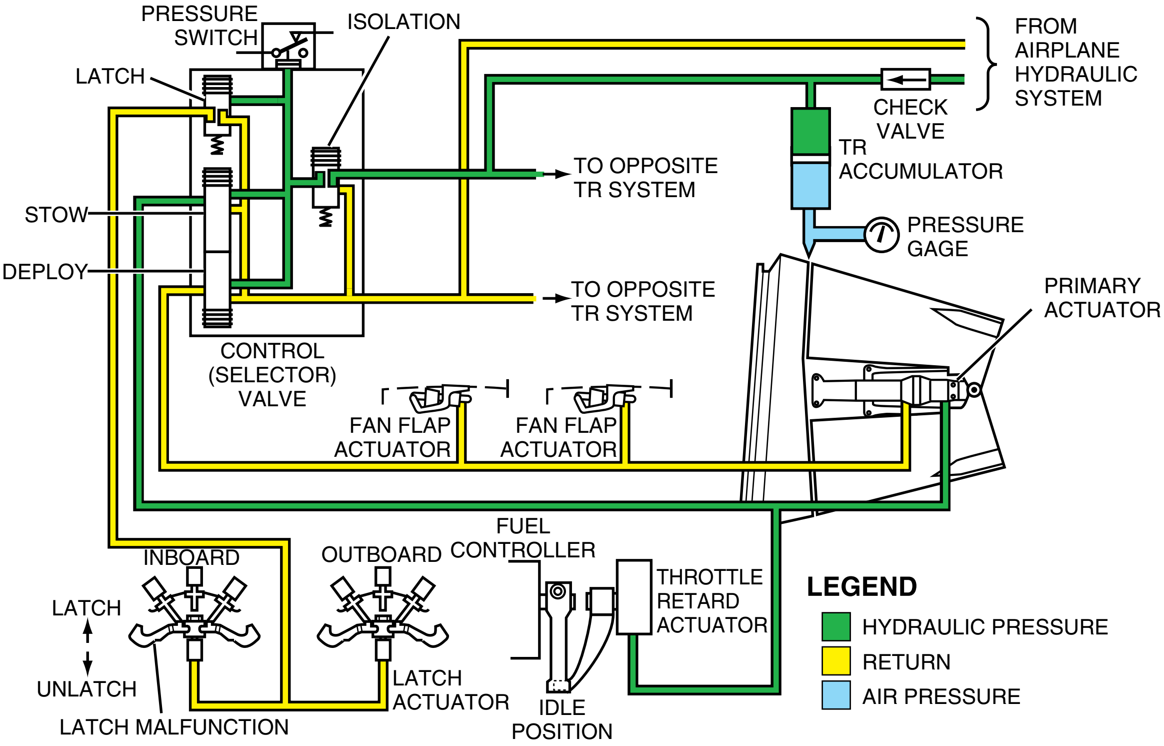


Figure 7-29. Autostow

AUTOMATIC THROTTLE RETARD

The throttle retard installation uses hydraulic pressure from the reverser system to force the engine fuel control mechanism toward idle when a stow signal is generated.

Should an inadvertent stow of the reverser occur while the reverser piggyback lever is in a reverser thrust range, the throttle retard mechanism automatically retards the reverser lever toward the reverse idle deploy position.

ELECTRICAL POWER

Electrical power to operate and control the thrust reverser is taken from the PE busbar.

NORMAL CONDITION MANAGEMENT

General

Deployment of the thrust reversers on unpaved surfaces is restricted to reverse idle thrust.

Reverser Door Pins

The reverser doors may be pinned in the open or closed positions. It is not permissible to pin only one door. The airplane may be operated with the doors secured in the closed position.

External Check

Before performing the external check, ensure that the OFF annunciators on the thrust reverser panel are illuminated. At each engine aft nacelle cowl, check that all access panels are securely fastened and that the thrust reverser clamshell doors are fully stowed. Ensure that safety pins are removed from the reverser doors. From the rear of the engine, check that fan flaps are retracted.

CAUTION

Reverser doors can be deployed without the engines operating. If the reverser doors are deployed, they must be stowed before engine start. One crewmember must observe outside while another operates the system.

Before Starting Engines

Press the FLT ANNUN TEST button, and ensure that the thrust reverser control panel annunciators illuminate. Ensure that the THRUST REVERSER POWER switch OFF annunciators are illuminated.

After Starting Engines

With the engines at idle, press the THRUST REVERSER POWER switch. Ensure that the ARM annunciators are illuminated and that the OFF annunciator is extinguished.

On each engine in turn, check reverse thrust operation as follows:

1. Ensure that the engine throttle levers are at idle. Ensure ARM annunciators are illuminated.
2. Advance right throttle lever until rudder bias is felt. Ensure that R/H ARM annunciator is extinguished.
3. Move the left thrust reverser lever to the deploy position, and check that the rudder bias pressure is removed from the pedals. Ensure that the L/H UNLCK and REVRS annunciators illuminate, and that the MWS REVERSER annunciator does not illuminate.
4. The left thrust reverser must be stowed and all appropriate indications valid before proceeding to test the right thrust reverser.

5. Ensure that the right throttle lever is at IDLE and that the R/H ARM annunciator is illuminated. Repeat the above steps using the right thrust reverser lever..

Landing

1. After touchdown, with the throttles at IDLE, lower the nosewheel to the runway.
2. Select lift dump, and use the wheel brakes in the normal manner.
3. Check that the ARM annunciators are illuminated.
4. Pull both thrust reverser levers up to the deploy position.
5. When the UNLCK and REVRS annunciators illuminate and the thrust reverser lever balks have released, the thrust reverser lever position may be selected as required.

NOTE

Reverse thrust must be reduced to idle by 50 knots.

Engine Shutdown

1. Before parking, place the thrust reverser levers to stow.
2. Check that the REVRS and UNLCK annunciators are extinguished.
3. Press the THRUST REVERSER POWER switch, and ensure that the OFF annunciator is illuminated.
4. Complete the engine shutdown procedures.

ABNORMAL CONDITION MANAGEMENT

Illumination of Thrust Reverser ARM or UNLCK Annunciator during Flight

If the ARM or UNLCK annunciators illuminate in flight, singly or in combination, the associated POWER switch(es) must be selected OFF.

Automatic Throttle Retard with Illumination of ARM, UNLCK, and MWS REVERSER Annunciators

If the illumination of the ARM and UNLCK annunciators is associated with the REVERSER annunciator and throttle retardation toward IDLE, the system has been armed automatically using the “autostow” cycle.

The associated POWER switch should be selected OFF. If indications revert to normal, the engine may be used as required. If these indications persist, leave the engine at idle for the remainder of the flight and land as soon as practicable. Do not use thrust reverse on the affected engine.

Illumination of MWS REVERSER Annunciator during Deploy

If the MWS REVERSER annunciator illuminated during the reverser deploy cycle, the rudder bias has not been inhibited. Power must not be increased above reverse idle.

LIMITATIONS

- Deployment of either thrust reverser is restricted to ground operations only.
- Engine starts with reversers deployed are prohibited.
- Use of reversers for power backing of the airplane is prohibited.
- Deployment of the thrust reversers on unpaved surfaces is restricted to reverse idle thrust.
- Full reverse minimum speed is 50 knots.

POWERPLANT LIMITATIONS

OPERATING LIMITS

LP Rotor RPM (N_1)

Takeoff	100%
Maximum continuous	100%

NOTE

Thermal binding of the low-pressure rotor may occur (because of differential cooling of the rotor components) if engine restarts are attempted between 25 to 45 minutes after engine shutdown. If an engine restart is to be attempted within the 25-45 minute time period, the fan should be rotated by hand through 90° or, alternatively, a motoring cycle should be carried out to ensure N_1 rotation. Particular attention should then be given to N_1 rotation during the subsequent start, and the start should be aborted if no N_1 rotation is indicated before 20% N_2 rpm.

HP Rotor Speed (N2)

Takeoff (normal)	100%
Takeoff (APR operating 5 minutes)	100.8%
Maximum continuous	100%

Interturbine Temperature (ITT)

Table 7-1 shows the rpm, ITT, and time limit parameters for five different engine operating conditions.

Oil Temperature

SL to 30,000 feet (maximum)	127° C
30,000 feet and above (maximum)	140° C
Transient (maximum)	149°C
Starting (minimum)	-40° C
Before takeoff, the minimum oil temperature	+30° C

Oil Pressure

Takeoff-climb and maximum continuous	38 to 46 psi
Transient (3 minutes maximum)	55 psi
Idle (no limit)	25 psi

Table 7-1. ENGINE OPERATING CONDITIONS

	% RPM		MAX	
CONDITION	N_1	N_2	ITT °C	TIME LIMIT
Starting and relighting	—	—	978° C	Unlimited time.
	—	—	996° C	10 seconds
		Over	996° C	5 seconds
			1,037° C	*
Normal takeoff (5 minutes)	100	100	978° C	5 minutes
Maximum (APR) takeoff, (5 minutes)	100	100.8	996° C	5 minutes
Maximum continuous	100	100	968° C	Unrestricted time, this is not a normal cruise setting.
Maximum overspeed	103	103	—	5 seconds
Maximum cruise	100	100	932° C	
Operating transients	—	—	1,006° C	5 seconds
	—	—	1,016 ° C	2 seconds
			**	
			***	

NOTE:

* If achieved or exceeded: record maximum temperature and duration in engine log. Hot section inspection must be carried out before further engine operation.

** 1,007 to 1,016° C for more than two seconds, or

*** 1,017° C attained or exceeded: enter maximum temperature and duration in engine log. Hot section inspection must be carried out before further engine operation.

POWERPLANT GROUND SERVICING

Engine Oil

Capacities

Total oil capacity of system:

U.S. pints—25

Imp. pints—22

Liters—12.5

Total capacity of tank:

U.S. pints—13.2

Imp. pints—11.7

Liters—6.24

NOTE

A card listing approved oil is located in the airplane manual stowage. The consumable quantity stated on the card refers to the oil available for use below the MIN level on the sight glass and is sufficient for more than 12 hours engine operation at maximum oil consumption.

Approved Engine Oils

Mobil Jet II

Esso 2380

Castrol 5000

Aero Shell 500

Mobil Jet 254

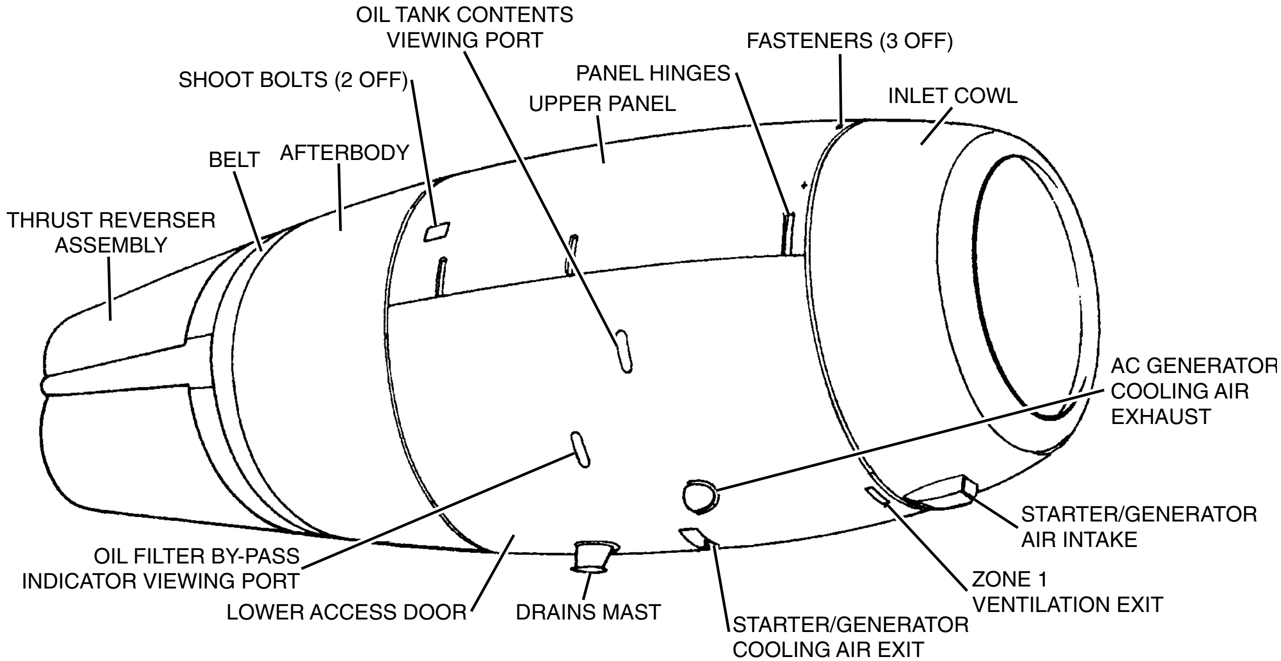
Aero Shell 560

Engine Oil System Servicing

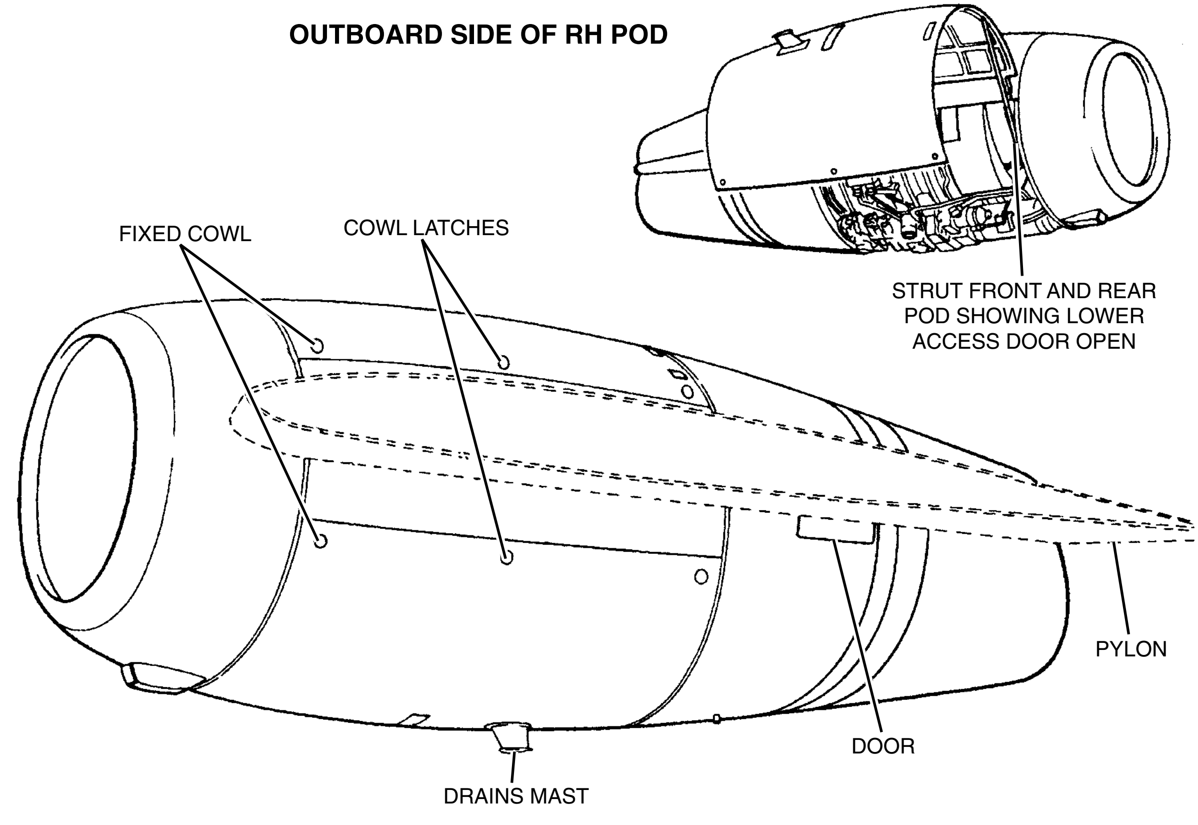
Check the level of the oil at the oil tank sight glass visible through the viewing port in the cowling lower access door.

NOTE

Check the oil level within 15 minutes of engine shutdown. If the oil level is not checked within this period, start the engine and run at idle for 5 minutes, then check the oil level. The oil level is indicated at the bottom of the sight glass float ball.

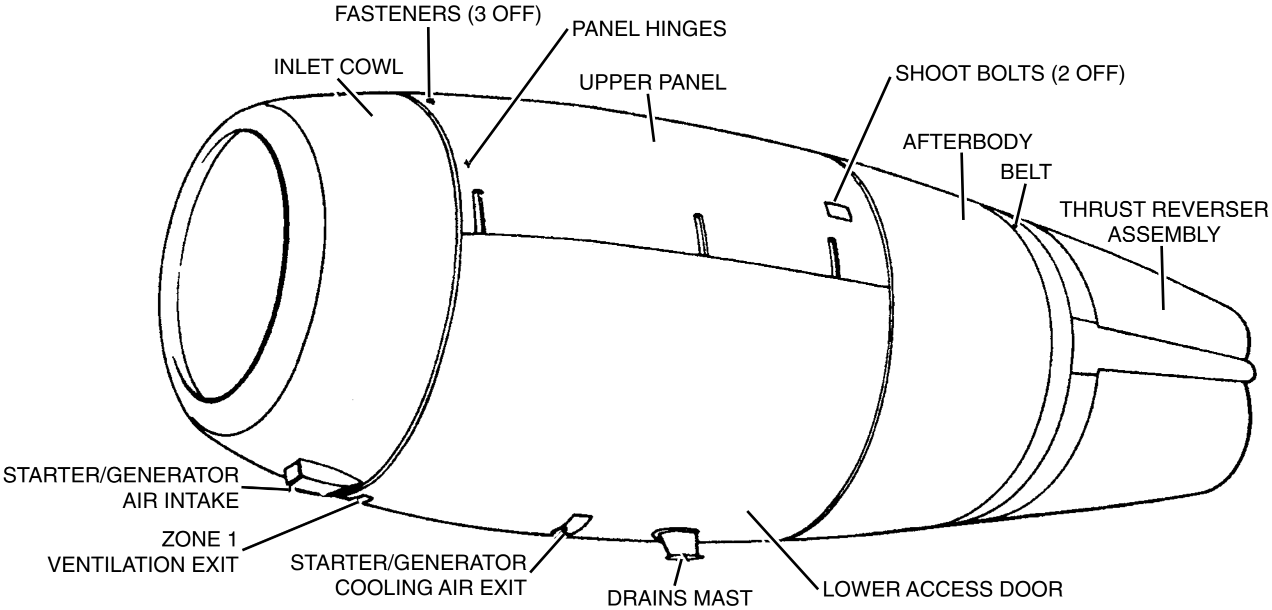

OIL TANK CONTENTS VIEWING PORT
FASTENERS (3 OFF)
SHOOT BOLTS (2 OFF)
PANEL HINGES
UPPER PANEL
INLET COWL
BELT
AFTERBODY
THRUST REVERSER ASSEMBLY
AC GENERATOR COOLING AIR EXHAUST
STARTER/GENERATOR AIR INTAKE
OIL FILTER BY-PASS INDICATOR VIEWING PORT
ZONE 1 VENTILATION EXIT
LOWER ACCESS DOOR
DRAINS MAST
STARTER/GENERATOR COOLING AIR EXIT

OUTBOARD SIDE OF RH POD

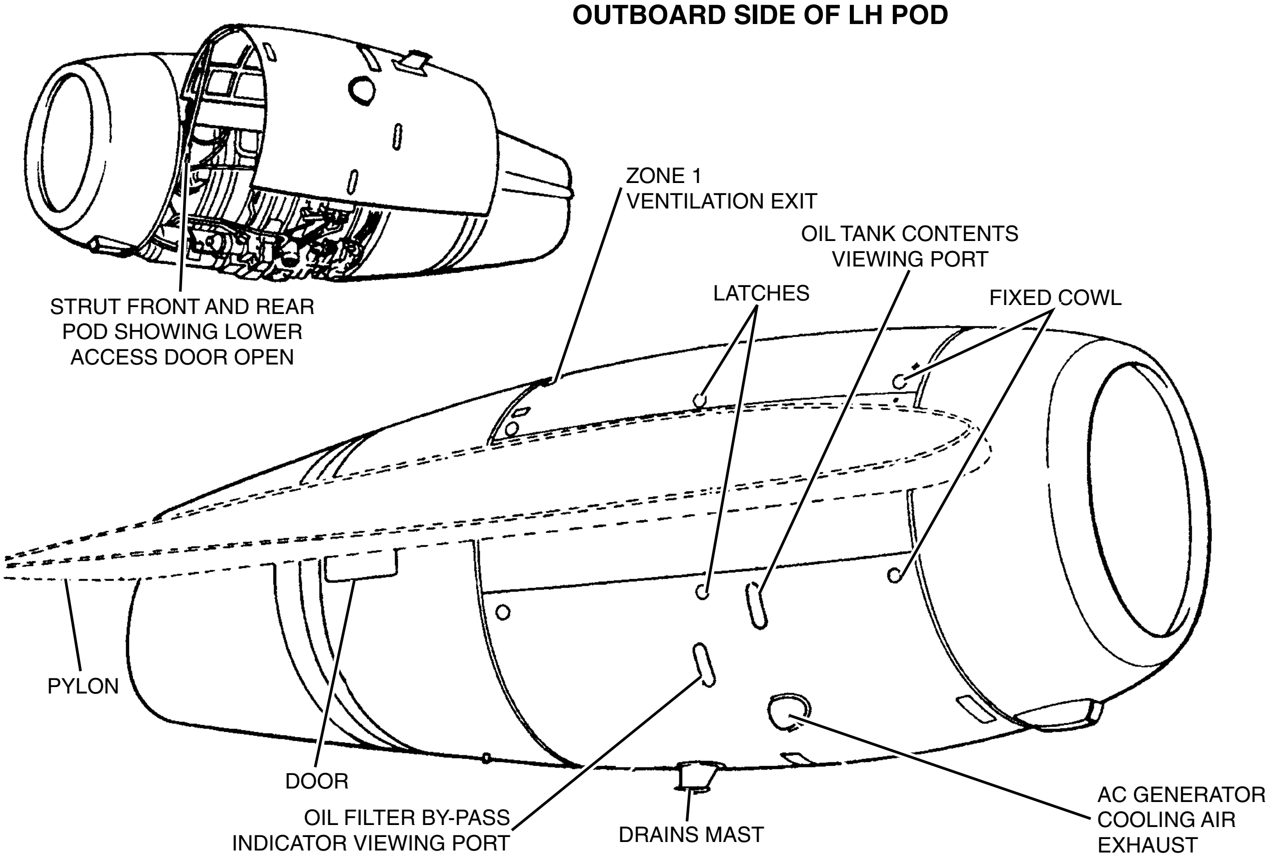


INBOARD SIDE OF RH POD

Figure 7-30. External View of No. 1 and No. 2 Engines (Sheet 1 of 2)



OUTBOARD SIDE OF LH POD


STRUT FRONT AND REAR POD SHOWING LOWER ACCESS DOOR OPEN
ZONE 1 VENTILATION EXIT
OIL TANK CONTENTS VIEWING PORT
LATCHES
FIXED COWL
PYLON
DOOR
OIL FILTER BY-PASS INDICATOR VIEWING PORT
DRAINS MAST
AC GENERATOR COOLING AIR EXHAUST

INBOARD SIDE OF LH POD

Figure 7-30. External View of No. 1 and No. 2 Engines (Sheet 2 of 2)

QUESTIONS

1. The powerplant classification on Hawker-800 XP series airplane is:
    A. Free-spool turbofan
    B. Reverse-flow turbofan
    C. Two-spool turbofan
    D. Single-spool turbojet

2. When the airplane is static on a standard day at sea level, each engine develops:
    A. 4,660 pounds of thrust
    B. 3,000 equivalent shaft horsepower
    C. 3,700 pounds of thrust
    D. 4,300 pounds of thrust at 98% N_2

3. The primary overspeed rpm-limiting device on the 731 series engine is the:
    A. N_1 ultimate overspeed sensor
    B. PT_2T_{T2} limiter
    C. N_2 ultimate overspeed sensor
    D. Mechanical governor in FCU

4. The maximum HP rotor (N_2) overspeed is:
    A. 100% for 5 minutes
    B. 103% for 5 seconds
    C. 105% continuous
    D. 101% momentary

5. The maximum oil pressure for power at or above idle rpm without a time limit is:
    A. 55 psi
    B. 25 psi
    C. 38 psi
    D. 46 psi

6. The maximum interturbine temperature for engine starting or relight is:
    A. 924° C over 30,000 feet for 10 seconds
    B. 952° C until stabilized at idle rpm
    C. 996° C for 10 seconds or 5 seconds in excess of 996° C
    D. 995° for the first 5 minutes of operation

7. The maximum transient oil pressure permissible is:
    A. 55 psi for 3 minutes
    B. 38 psi for takeoff, climb, and maximum continuous
    C. 55 psi for all operations under 25,000 feet
    D. 25 psi for a maximum of 3 minutes

8. If the MWS reverser annunciator illuminates during the deploy cycle it indicates:
    A. T/R’s are not deployed
    B. Rudder bias not inhibited
    C. Throttles are above idle
    D. Auto stow cycle activated

9. To facilitate an accurate setting of climb power:
    A. MWS annunciator comes on
    B. APR light goes out
    C. A green light in ITT indicator lights
    D. APR green light illuminates

10. The APR system is available:
    A. To both engines at all times
    B. If the engine sync system is on
    C. When both computers are functioning
    D. To the engine with an operative computer

CHAPTER 8
FIRE PROTECTION

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  Rear Equipment Bay Overheat 8-3

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ELECTRICAL POWER SOURCES 8-6

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8-2	Rear Equipment Bay Detection	**8-3**
8-3	Engine Fire-Extinguishing System	**8-4**
8-4	Pressure Relief Discharge Indicators	**8-4**
8-5	APU Extinguisher System	**8-5**
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CHAPTER 8
FIRE PROTECTION

INTRODUCTION

The Hawker 800 XP fire protection system includes the detection and warning of fire and overheat conditions in each engine and within the auxiliary power unit (APU). Overheat detection capability for the rear equipment bay is also provided. Fire extinguishers are installed to extinguish engine and APU fires. A portable hand-held extinguisher is located within the cockpit for extinguishing a fire inside the fuselage.

GENERAL

The engine fire protection system is composed of two sensing elements and responders per engine, a warning indicator on the roof panel for each engine, a warning bell, two fire extinguisher bottles which are activated from the cockpit, and two fire detection circuit test switches. The engine fire extinguishing system is a two-shot system; if an engine fire is not extinguished with actuation of the first bottle, the second bottle is available for discharge into the same engine. The engine fire bottles are located in the rear equipment bay.

Exterior pressure relief discharge indicators provide a visual indication if either fire bottle has been discharged by thermal expansion.

The fire protection system for the APU includes three overheat detector switches, a warning indicator light on the APU control panel, the fire bell, one fire extinguisher bottle which is activated from the cockpit, and a FIRE detection circuit test switch.

Overheat detectors are installed in the rear equipment bay. An indicator light on the MWS panel is provided to warn of overheat conditions.

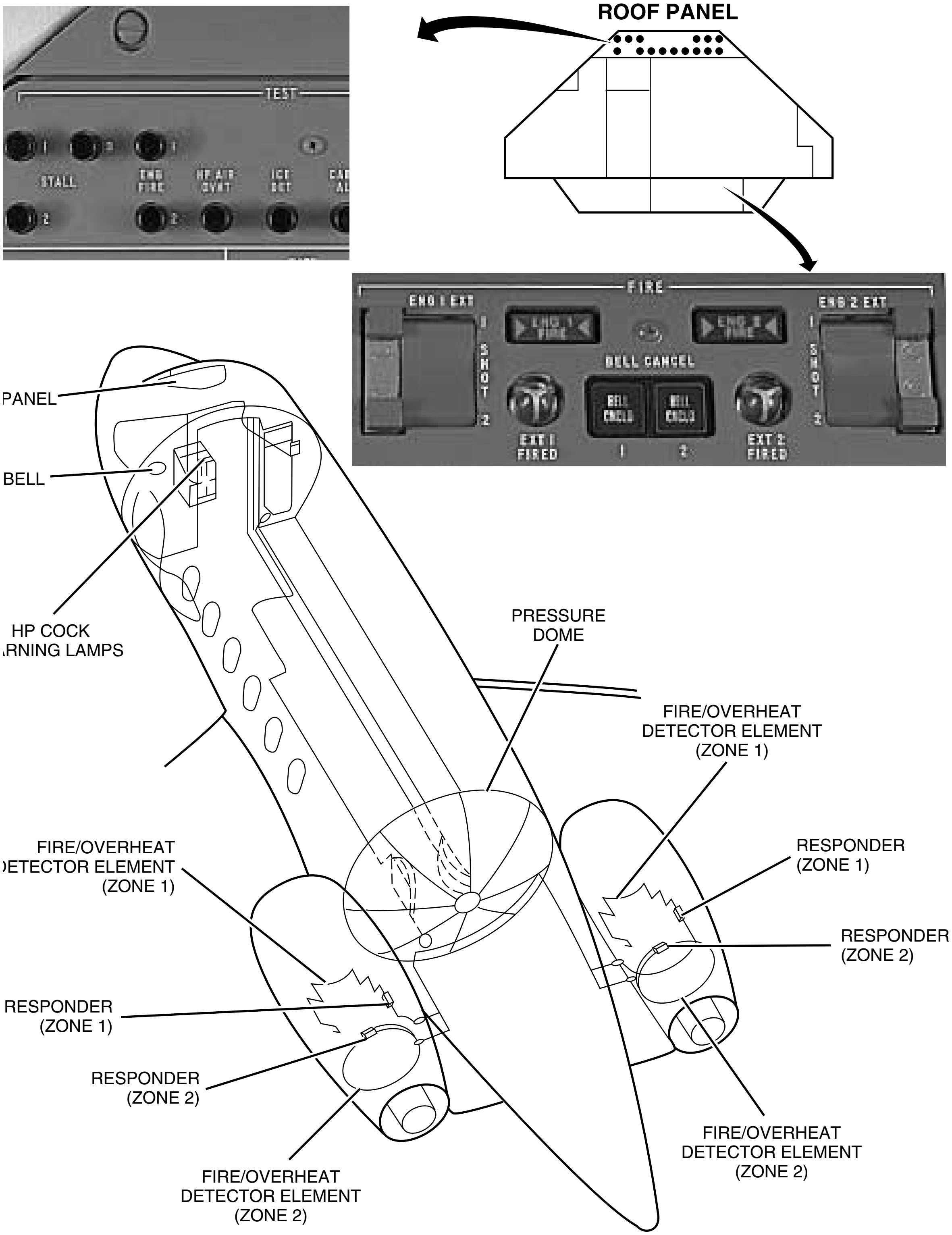

Figure 8-1. Fire Detection and Warning System

FIRE DETECTION

ENGINE FIRE/OVERHEAT

Fire or overheat in each engine is detected by two fire/overheat responders connected to a continuous-wire sensor element (Figure 8-1). The sensor element is a sealed tube containing gas. The gas expands as the temperature increases, and, at a predetermined temperature, the expanding gas initiates visual and audible warnings in the cockpit. As cooling occurs, the contraction of the gas in the element cancels the warnings.

The detector system in each engine sends a signal to its associated ENG 1 or 2 FIRE annunciator on the FIRE panel and to the respective repeater annunciators located on the MWS panel. Also illuminated by an engine fire/overheat detection signal are the red MWS flashing annunciators and the respective HP cock warning light.

The fire detection signal also operates the fire warning bell. Bell operation can be silenced by depressing the appropriate BELL CANCEL switchlight which then illuminates BELL CNCLD in white letters. These annunciators are located on the FIRE panel which is part of the roof panel (see Figure 8-1).

Each detector circuit has a TEST button, also located on the roof panel. Pressing either TEST switch illuminates the appropriate ENG FIRE annunciator on the FIRE panel and the respective ENG FIRE repeater on the MWS panel, causes both red MWS annunciators to flash, lights the associated HP cock warning light, and rings the fire warning bell.

The fire detection system in each engine will not test unless both fire detector elements, in the engine being tested, have the proper pressure levels.

All annunciators are shown in the Annunciator Chapter.

REAR EQUIPMENT BAY OVERHEAT

Six thermal switches are located in the rear equipment bay. If one or more sense an overheat condition, a signal is sent to the red REAR BAY OVHT annunciator on the MWS center panel and also to both MWS red flashers. Figure 8-2 shows the location of the rear bay overheat detector switches.

FIRE EXTINGUISHING

ENGINE FIRE EXTINGUISHING

Two dual-head, single-shot fire extinguisher bottles are located in the rear equipment bay. Each bottle is capable of discharging its total store of agent into zone 1 of either engine. Therefore, the pilot can select both extinguishers for either engine or a single shot for each engine.

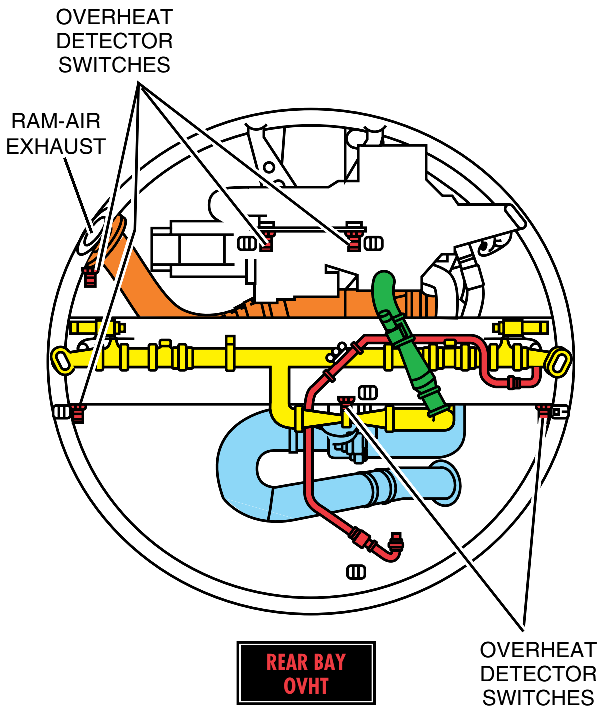

Figure 8-2. Rear Equipment Bay Detection

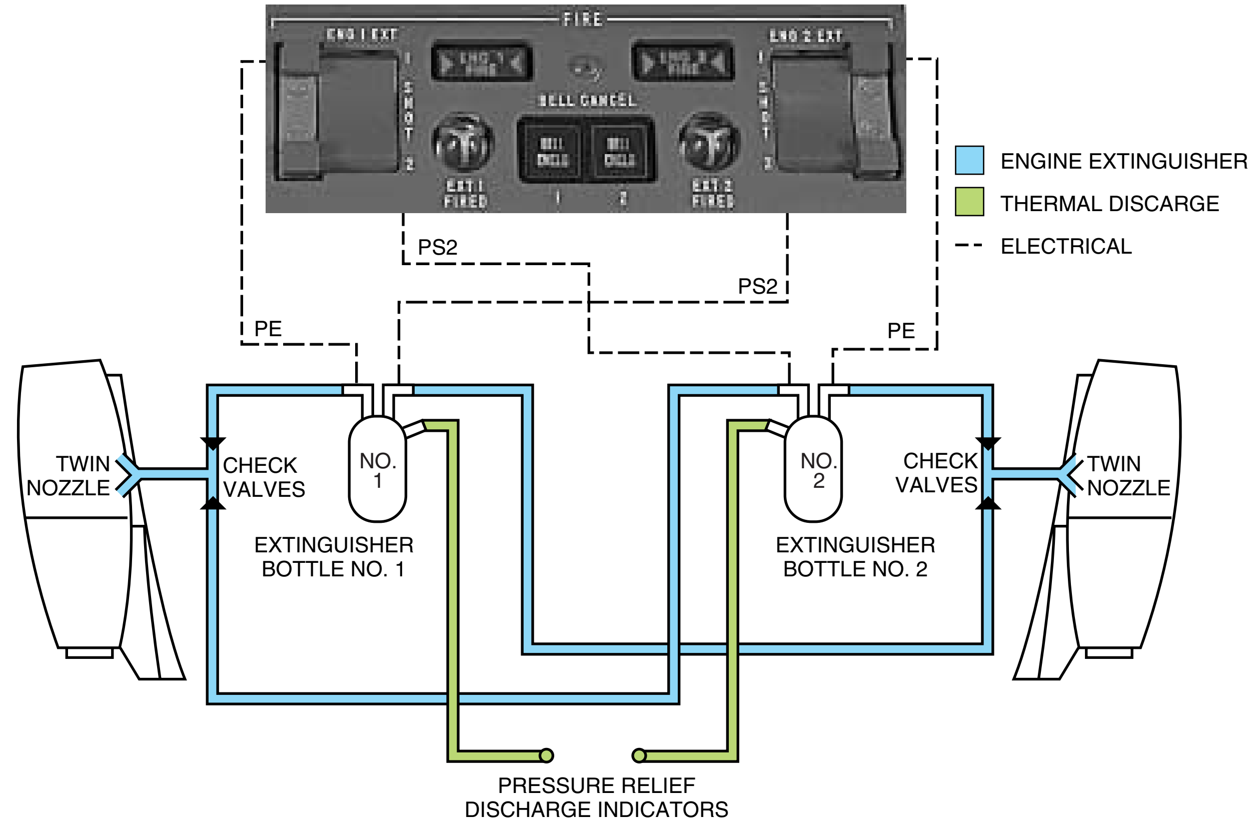

Figure 8-3. Engine Fire-Extinguishing System

Each bottle discharge is achieved by the activation of electrically operated units. Either pilot can select which extinguisher is to be fired into which engine by properly activating the appropriate ENG 1 or 2 EXT switch to SHOT 1 (up) for its respective engine, or down to SHOT 2 for the same engine (Figure 8-3). When one of the extinguishers has been fired, the associated EXT 1 or 2 FIRED indicator turns red. Power for both SHOT 2 circuits is supplied by bus bar PS2. All other fire protective circuits are powered by bus bar PE.

Figure 8-4 shows the pressure relief discharge indicators which are located under the left engine pylon on the fuselage. They are green discs. A buildup of excess thermal pressure in the bottle will cause the extinguishing agent to discharge, blowing out the disc and leaving a red indication. The condition of the discs should be checked during the exterior preflight inspection.

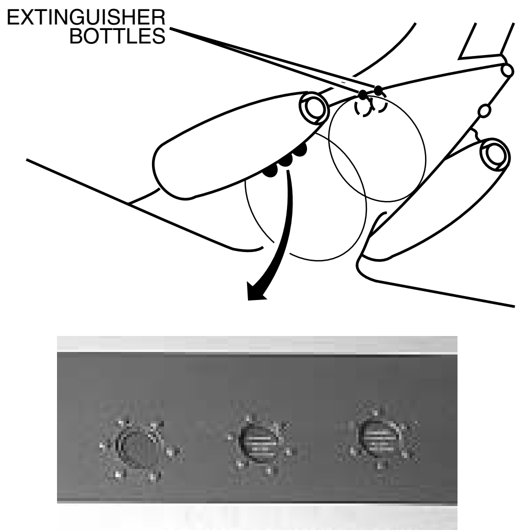

Figure 8-4. Pressure Relief Discharge Indicators

APU FIRE EXTINGUISHING

The APU has its own fire extinguisher (Figure 8-5). Discharging of the APU fire bottle is controlled by a guarded switch on the APU control panel (see Figure 8-6) that fires the cartridge on the single-shot bottle which is located in the rear equipment bay. In addition to discharging the bottle, the switch also isolates the DC power to the APU fuel solenoid valve, ensuring APU shutdown.

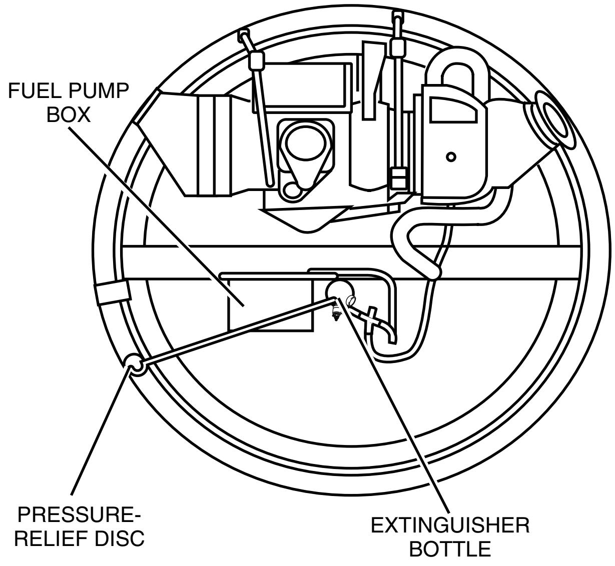

Figure 8-5. APU Extinguisher System

This bottle has its own red pressure relief indicator disc, located on the exterior of the airplane under the left engine pylon. If the bottle is discharged due to excessive heat buildup, it will blow out the red disc, leaving a red indication.

APU FIRE DETECTION

The APU fire detection system comprises three overheat detector switches, a red FIRE warning light, and the fire bell.

If an overheat condition is detected by one or more of the detector switches, a signal is transmitted to illuminate the FIRE warning indicator, ring the fire bell, and shut down the APU. The fire bell can be silenced by moving the master switch to OFF (Figure 8-6), but the FIRE light remains on until the fire has been extinguished or the overheat condition remedied. The momentary TEST button, located adjacent to the FIRE light, tests the light and the bell.

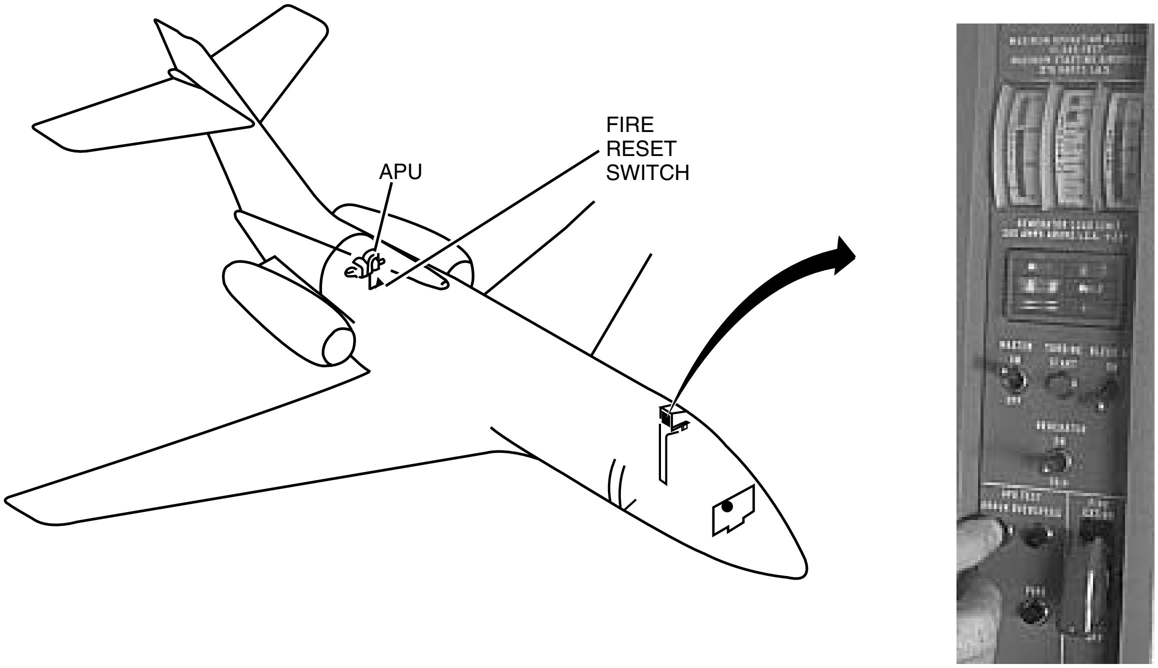

Figure 8-6. APU Fire Detection

PORTABLE (FUSELAGE) FIRE EXTINGUISHING

A portable fire extinguisher is stowed on the left bulkhead within the cockpit (Figure 8-7). It is provided for fighting fires within the cockpit and cabin. The contents of the extinguisher can be discharged intermittently or continuously by depressing the operating lever. A guard, embodied in the stowage bracket, prevents operation of the lever when the extinguisher is stowed. A toggle clamp must be released before the extinguisher can be removed from the bracket.

The extinguisher is fitted with a FULL indicator disc. When the extinguisher is operated, the disc ejects as soon as discharge commences.

The extinguisher is suitable for fighting fires which are fueled by flammable liquids, electrical sources, or other sources such as wood, paper, and cloth.

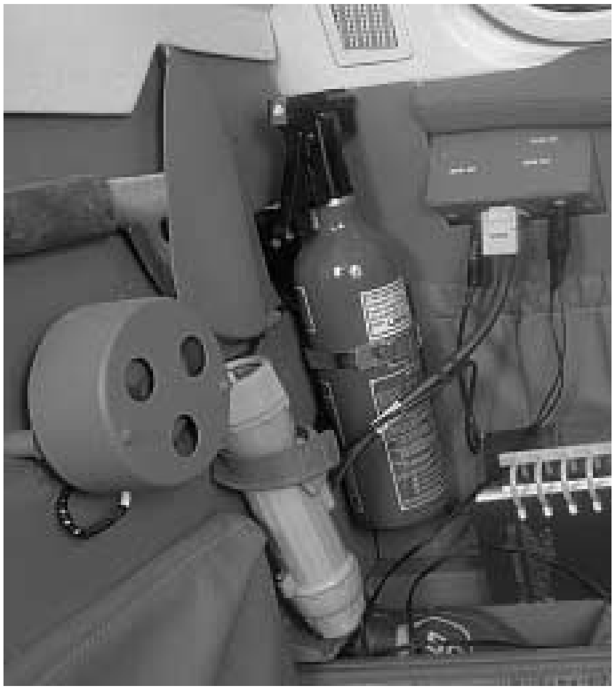

Figure 8-7. Portable Fire Extinguisher

ELECTRICAL POWER SOURCES

PE bus bar

- Fire protection circuits
- Shot 1 (both engines)
- APU

PS2 bus bar

- Shot 2 (both engines)

QUESTIONS

1. Engine fire extinguisher bottles are located in the:

A. Nacelles
B. Engine pylons
C. Rear equipment bay
D. Baggage compartment

2. The exterior preflight check of the fire extinguisher includes:

A. Checking the condition of one engine and one APU discharge indicator
B. Checking the condition of one engine and two APU discharge indicators
C. Checking the condition of two engines and one APU discharge indicator
D. Checking the engine fire extinguisher pressure gages

3. The engine fire bell can be silenced:

A. Only during the test procedure
B. Only when the fire has been extinguished
C. By pressing the BELL CANCEL switch
D. By turning off the APU MASTER switch

4. The engine fire bottles are discharged as follows:

A. Each bottle can be used only for its respective engine.
B. When a visual fire warning appears, discharge SHOT 1 first. If necessary, then discharge SHOT 2.
C. Each bottle can deliver two individual shots, so a total of four are available.
D. They are usable only if the pressure relief discharge indicators appear red.

5. For a fire in the APU:

A. A switch on the APU control panel controls the firing of a separate fire extinguisher.
B. Either of the engine fire extinguisher bottles may be used for an APU fire.
C. There will be fire-warning indication, but the switch only shuts down the fuel to the APU-there is no extinguishing agent.
D. A crewmember must take the portable hand-held extinguisher to the rear equipment bay.

6. The following bus bar(s) provide(s) DC power for engine fire extinguisher operation:

A. PE and PS1
B. PS1
C. PS2
D. PE and PS2

CHAPTER 9
PNEUMATICS

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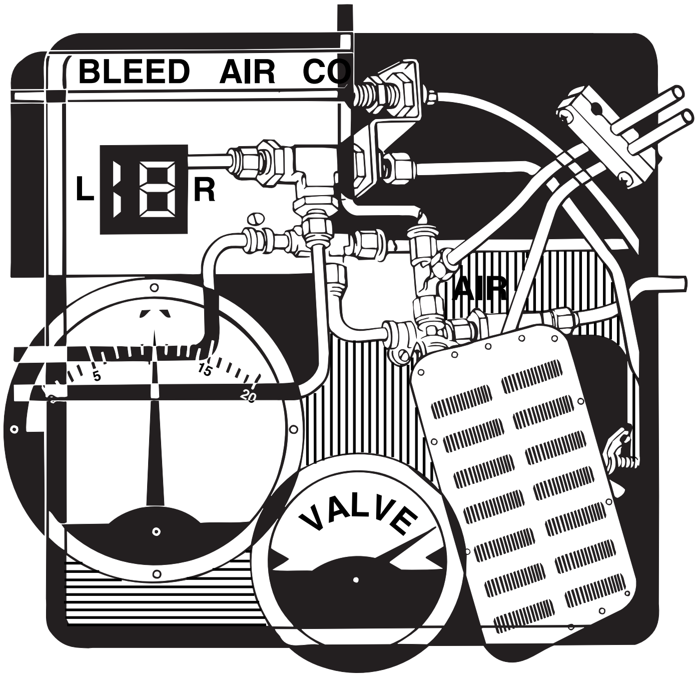
9-2 Pheumatic System Control Diagram **9-3**

9-3 Main Air Valve Switch LP On - Selected—
HP Valve and Mixing Valve Closed **9-4**

9-4 Main Air Valve Switch Open - Selected—
HP Valve Mixing Valve Open **9-5**

9-5 Flood Air **9-6**

CHAPTER 9
PNEUMATICS



INTRODUCTION

The Hawker 800 XP airplane incorporates a pneumatic system used for supplying engine bleed air to the air-conditioning system, rudder bias system, hydraulic reservoir pressurization, and the pressurization system. The pneumatic system includes several valves, a pressure regulator and shutoff valve, and operating controls. It also supplies bleed air to other airplane systems.

DESCRIPTION AND OPERATION

GENERAL

The pneumatic system receives low-pressure (LP) and high-pressure (HP) air from both engines when they are running. The bleed air provides the air supply used by other dependent systems, which are:

- Air-conditioning system
- Pressurization system
- Rudder bias system
- Hydraulic reservoir pressurization

HP bleed air supplements the LP bleed air at the mixing valve whenever the bleed-air pressure drops to 27 psi, with the MAIN AIR VLV switches selected to OPEN. This condition normally occurs during descent of the airplane when the engine rpm is reduced. When the LP bleed air drops to 27 psi, a pressure switch causes the HP bleed-air valve to open, allowing HP air to enter the mixing valve. A temperature limiter in the mixer valve operates at 200° C (nominal) to reduce the HP air bleed. Thus, the LP bleed air is supplemented during the airplane's descent or at any other time that the LP bleed-air pressure drops below 27 psi. The supplemental HP bleed air keeps the pressure above 27 psi for operation of the dependent systems.

From each main air valve, the air is ducted via a nonreturn valve, and then via a common duct to a pressure regulator and shut-off valve unit (PRSOV), and a pressure reducing and flow restricting venturi to an air cycle machine. (Refer to Chapter 11, "Air Conditioning."). Bleed air from the No. 2 engine may be used for auxiliary heating or emergency pressurization.

CONTROLS AND INDICATORS

The pneumatic system's controls consist of two MAIN AIR VLV 1 and 2 switches. Each switch has three positions labeled "OPEN," "LP ON," and "CLOSE" and is located on the roof panel. (See Figure 9-1.)

Engine Air Bleeds

Bleed air from each engine enters the rear bay via two ducts in each pylon. Low pressure (LP) air is ducted from a flow limiting venturi to a mixing valve.

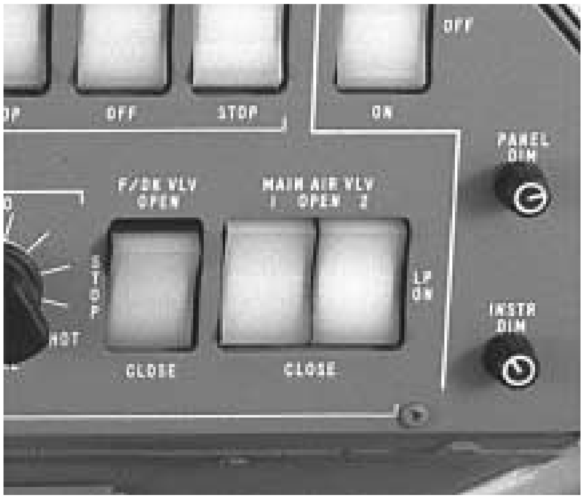


Figure 9-1. Pneumatic System Control Switches

High pressure (HP) air, when both main air valves (MAVs) are selected open, is ducted from a flow limiting venturi to a solenoid-operated shutoff valve then to the mixing valve.

Mixing Valve

The mixing valve regulates HP air (when it is demanded by LP air). It mixes the HP and LP air supplies, maintaining a minimum pressure of 18 psi at the valve outlet and limiting the temperature to approximately 200°C.

The amount of HP air used is dependent on the LP air pressure available. When the LP air pressure falls to 27 psi, the HP valve is opened but the mixing valve does not mix any HP air into the system until the LP air pressure falls to 18 to 22 psi and below.

Main Air Valve (MAV)

Air from each mixing valve is fed through an electrically actuated main air valve (MAV) and a nonreturn valve (NRV) into a common supply duct. The MAV and HP valve selections are made by means of two switches, MAIN AIR VLV 1 and MAIN AIR VLV 2, on the flight compartment roof panel (Figure 9-1).

The switch selections of CLOSE, LP ON, and OPEN affect the system as follows:

- CLOSE: MAV and HP valve are both selected closed.
- LP ON: MAV selected open and HP valve selected closed.
- OPEN: MAV selected open. HP valve 'armed' but the valve position is controlled by the LP duct pressure switch.

Time delay units are incorporated in the "open" circuit of the MAVs to prevent an initial surge of air to the cabin. The No. 1 MAV has a staggered open/delay logic control, and takes approximately 20 seconds to reach the fully open position. The No. 2 MAV is fast acting but is delayed by 30 seconds before running fully open in approximately one second.

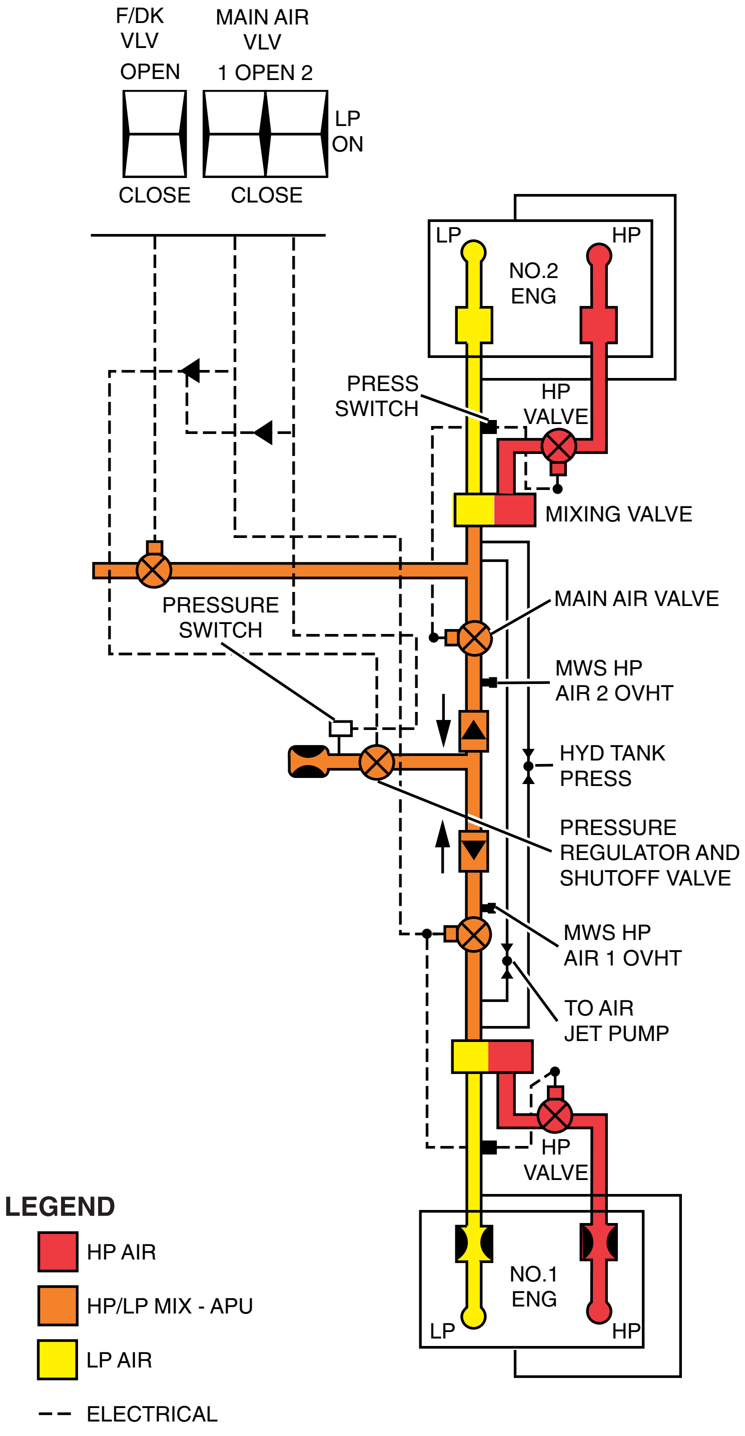


Figure 9-2. Pneumatic System Control Diagram

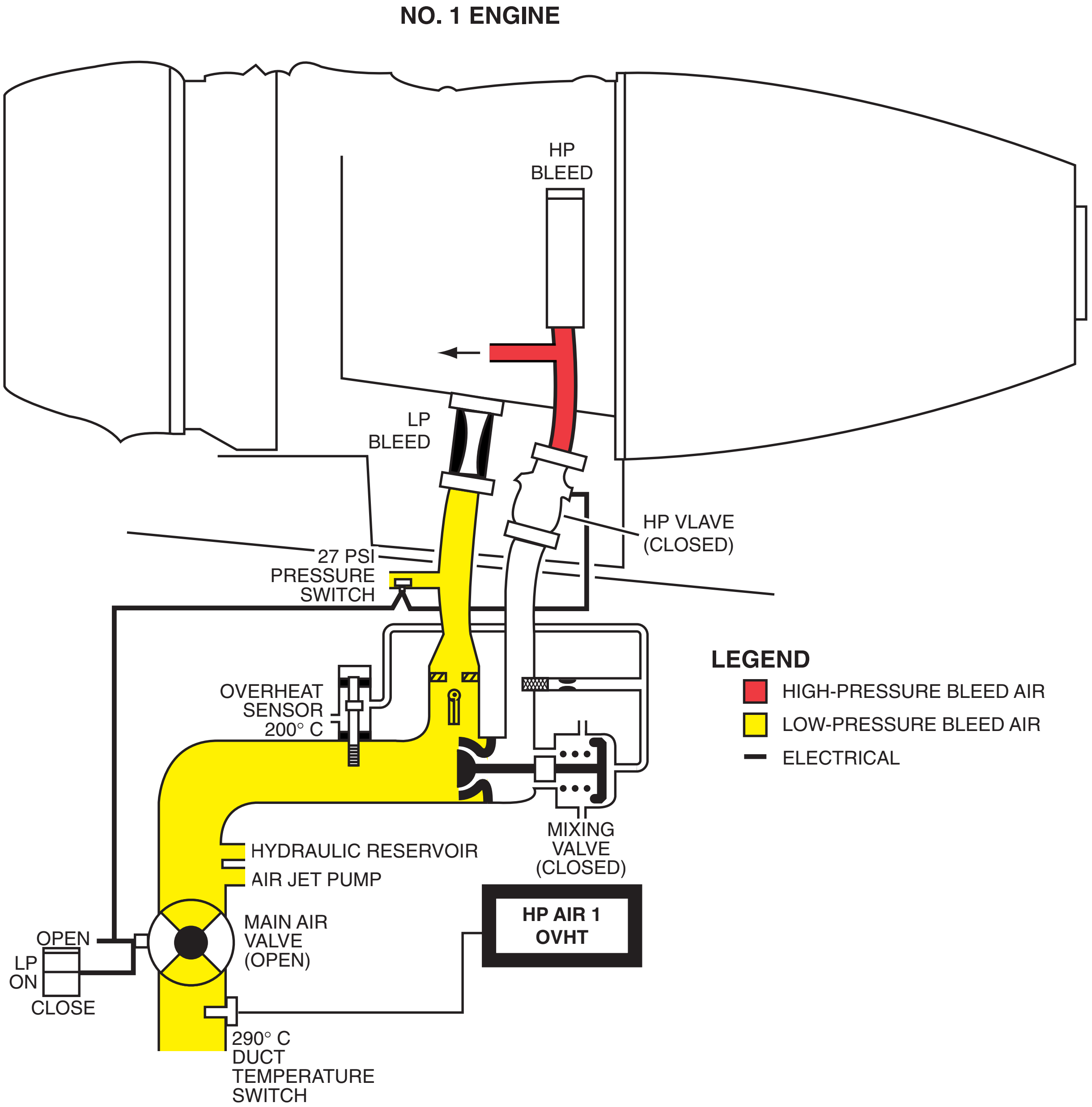


Figure 9-3. Main Air Valve Switch LP On – Selected—HP Valve and Mixing Valve Closed

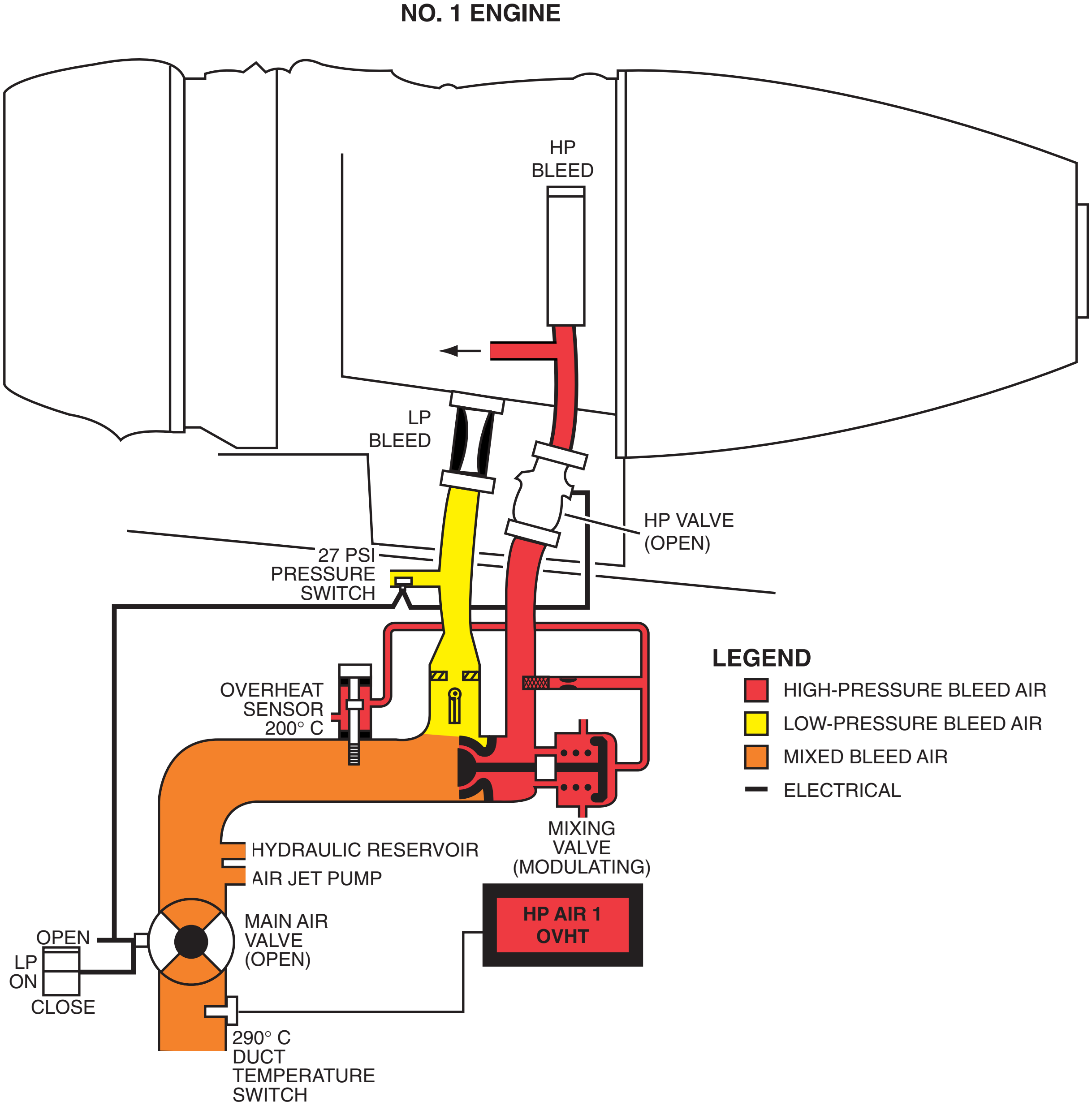


Figure 9-4. Main Air Valve Switch Open – Selected—HP Valve and Mixing Valve Open

Flood Air Control

Temperature controlled air enters the pressure cabin through a silencer, a non-return valve and an electrically operated flood valve, which is controlled by a FLOOD OPEN/CLOSE switch on the flight compartment roof panel.

With FLOOD OPEN selected, conditioned air is supplied to the cabin flood flow outlet, at roof level at the right rear of the passenger cabin.

With FLOOD CLOSE selected, the air supply to the cabin flood flow outlet is cut-off and temperature controlled air is supplied to the cabin high level outlets and the flight compartment duct.

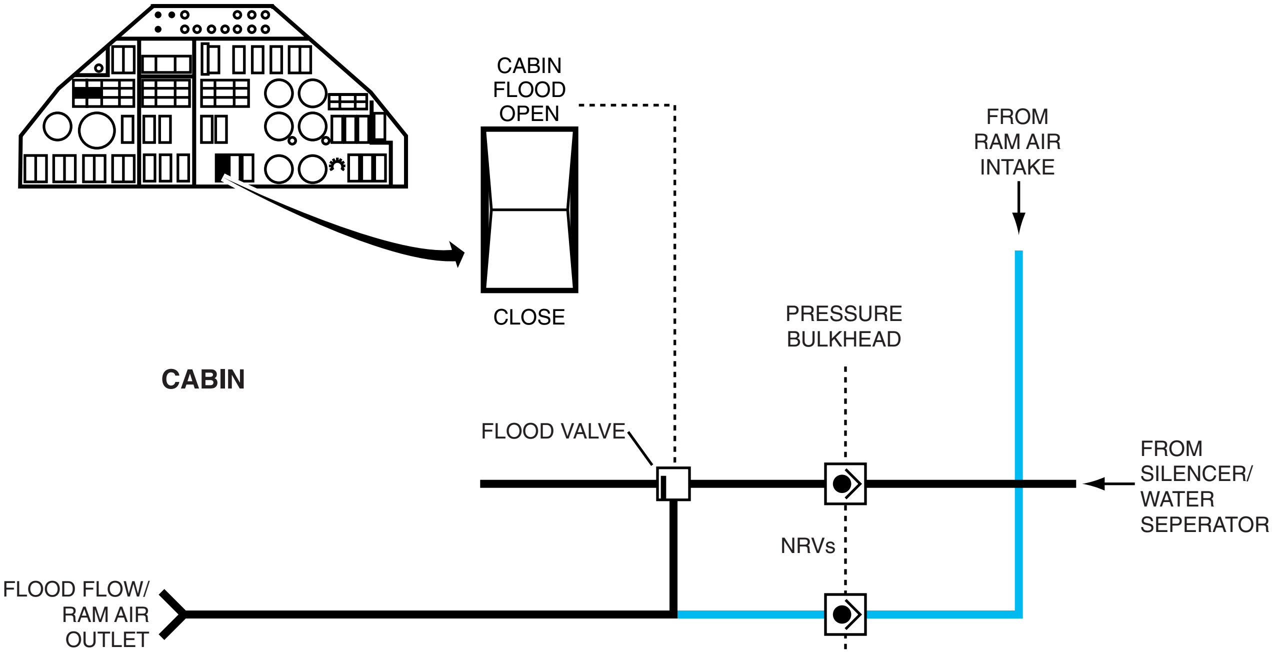
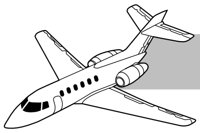


Figure 9-5. Flood Air



ELECTRICAL POWER SOURCES

DC POWER

PE bus bar

- HP AIR OVHT annunciators
- MAIN AIR VALVE 1 and 2 annunciators
- HP bleed on-off valve

LIMITATIONS

Main air valves and flight deck heat valve must be closed prior to takeoff and landing.

QUESTIONS

1. The HP bleed air supplements the LP bleed air at the mixing valve when the MAIN AIR VLV switch is selected to OPEN and:

A. LP bleed-air pressure drops to 27 psi
B. HP bleed-air pressure drops to 25 psi
C. LP bleed air rises above 25 psi
D. None of the above

2. The MAIN AIR VALVE 1 amber annunciator illuminates in flight if opening of the No. 1 air valve takes more than:

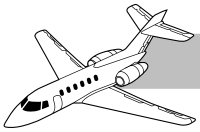
A. 15 seconds
B. 10 seconds
C. 50 seconds
D. 12 or 45 seconds

3. A pressure switch controls the positioning of an HP bleed-air valve. The position of the MAIN AIR VLV 1 or 2 switch in which electrical power is applied to a pressure switch is:

A. HP ON
B. LP ON
C. AUTO
D. OPEN

4. The LP ON position of the MAIN AIR VLV switches is selected:

A. Below 35,000 feet
B. If system failure occurs
C. A and B
D. None of the above



CHAPTER 10
ICE AND RAIN PROTECTION

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CHAPTER 10
ICE AND RAIN PROTECTION

INTRODUCTION

The Hawker 800 XP is approved for flight in known icing conditions. The systems are designed to allow safe flight through continuous maximum icing and intermittent maximum icing as defined in FAR Part 25 with one engine inoperative. Although intended primarily for in-flight operation, some of the systems may be used for ground operation.

GENERAL

The ice detection system consists of a rotary-cutter ice detector on the left side of the nose section and ice lights installed in the left and right wing fairing for each wing.

The airframe deicing system uses TKS fluid to anti-ice the leading edges of the wings and horizontal stabilizers.

The left and right windshields and forward side screens are electrically heated for anti-icing.

The pitot heads, forward static plates, rudder bias struts, and airflow angle sensors are electrically heated to prevent ice formation.

Each engine nacelle intake cowling is anti-iced by engine HP bleed air. The P_2T_2 sensor is electrically heated.

Figures 10-1 and 10-2 show the weather-protected airplane surfaces and ice protection components.

ICE DETECTION SYSTEM

The ice detection system consists of a rotary-cutter ice detector mounted on the left side of the aft nose section and ice lights for the left and right wings. The roof panel houses a test button, ICE DET switch, and annunciators (Figure 10-3).

The ice detector is controlled by the ICE DET switch on the roof panel and the left main gear squat switch. Selecting the ICE DET switch to AUTO inhibits detector operation on the ground and automatically turns the detector on as the left main gear strut extends during takeoff. Setting the switch to OVRD bypasses the left main gear squat switch and powers the detector circuit both on the ground and in flight.

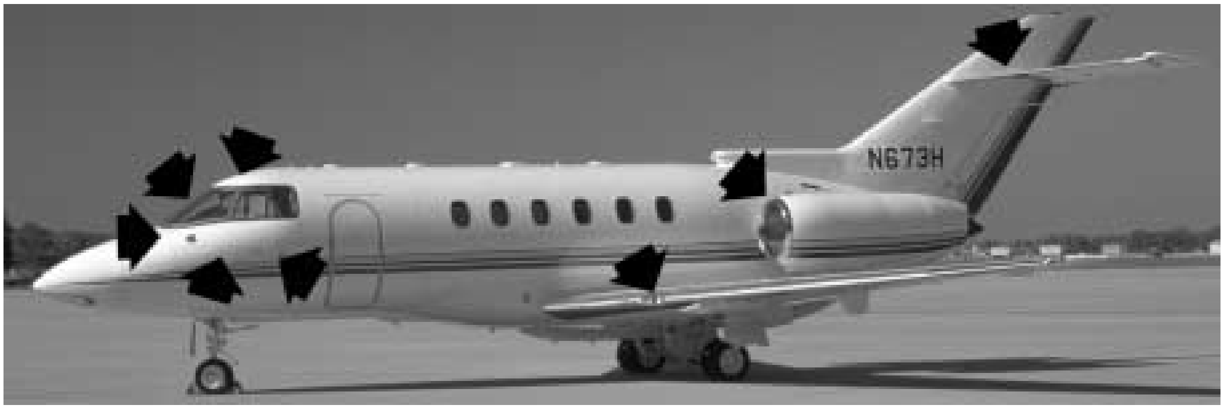


Figure 10-1. Weather-Protected Airplane Surfaces

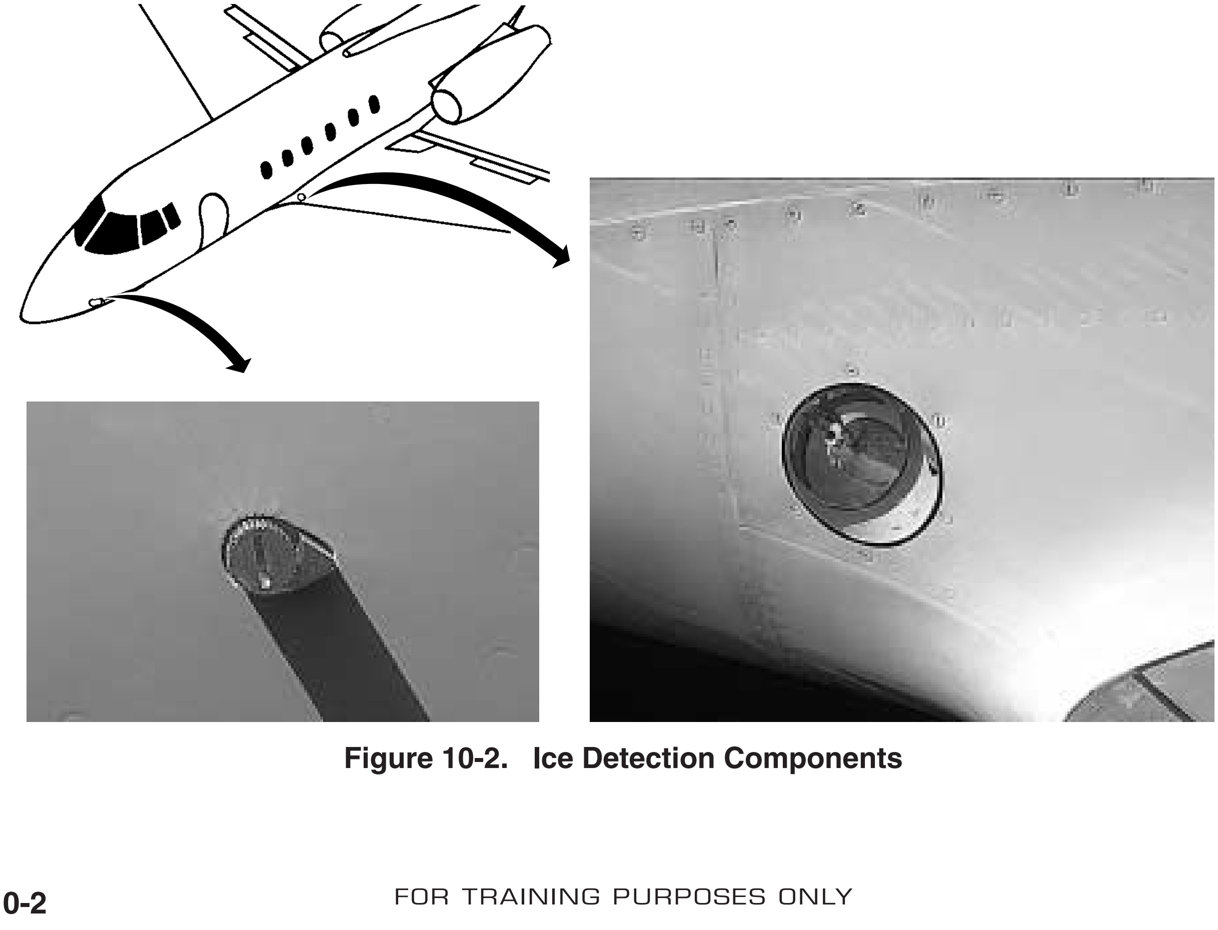


Figure 10-2. Ice Detection Components

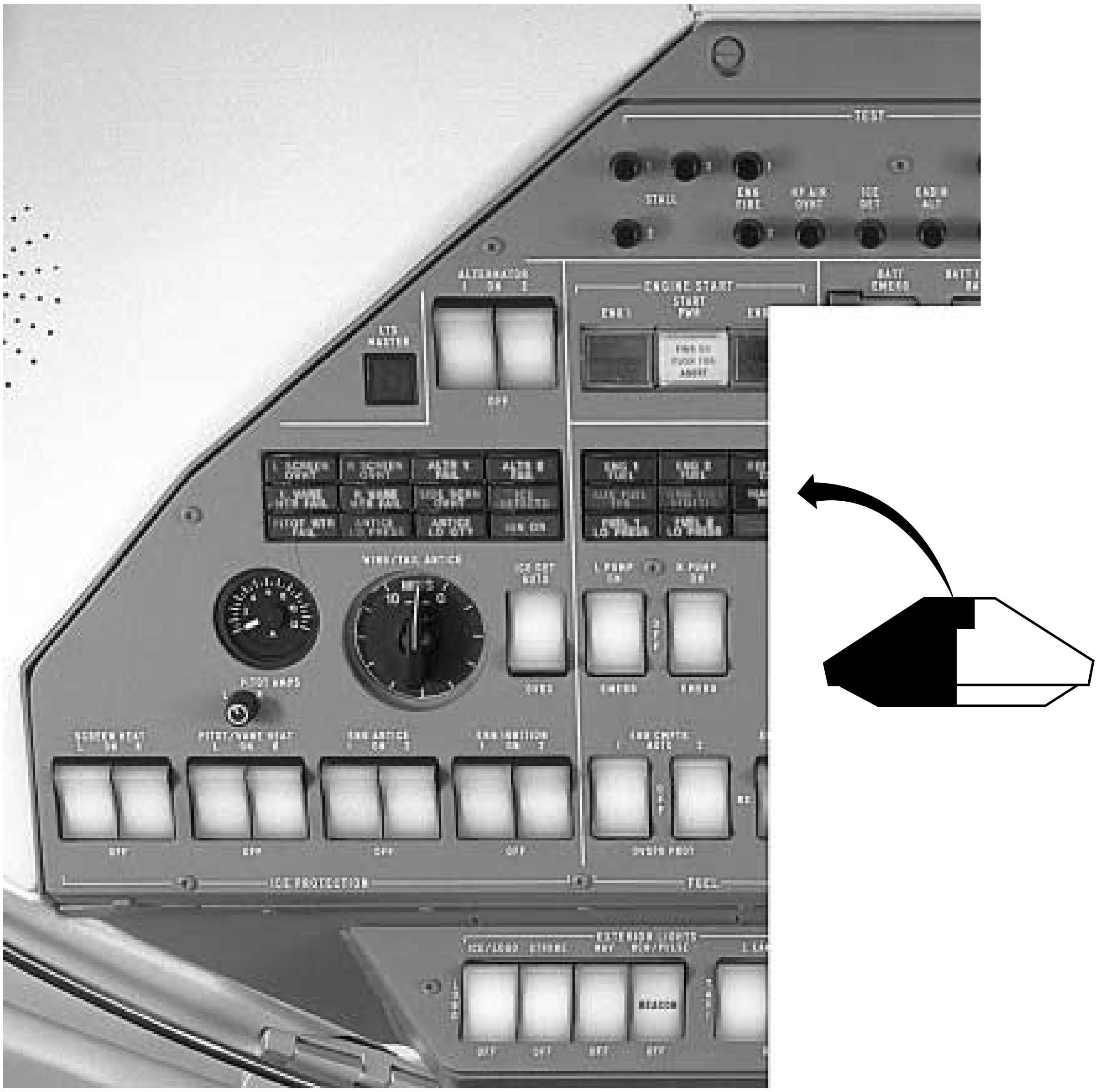


Figure 10-3. Ice Protection Controls

The ice detector is powered from the XS2 bus bar. An AC-powered motor drives a serrated rotor which rotates close to a knife-edge cutter. When ice forms on the rotor, the gap between the rotor and adjacent cutter is filled. The shaving action of the cutter against the ice increases motor torque, causing the motor to rotate slightly within its mounting. The rotation activates a microswitch which connects a DC power supply and illuminates the amber ICE DETECTD annunciator on the roof panel and the amber ICE PROT ⭡ annunciator light on the MWS panel. When icing ceases, the microswitch opens and, after a 30-second delay, the annunciators extinguish. The ICE DET pushbutton switch in the roof panel TEST group energizes both lights.

NOTE

The 30-second delay also occurs during system testing.

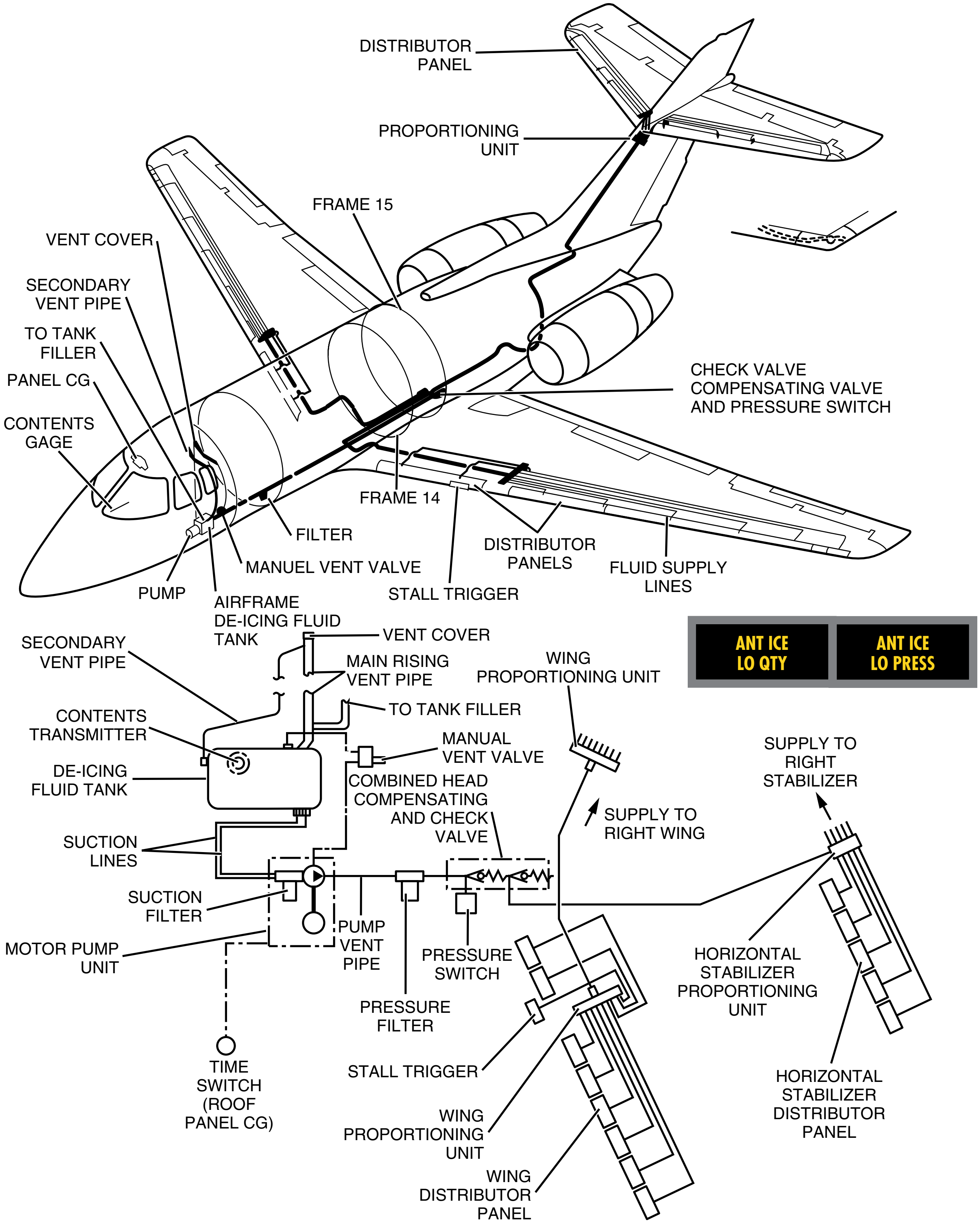


Figure 10-4. Airframe Anti-icing

The ice lights illuminate the leading edge of both wings for visual inspection at night. These ice lights are controlled by the ICE/LOGO switch in the roof panel EXTERIOR LIGHTS group. Power for the wing inspection lamps is from the PS1 bus bar.

The ice warning annunciators are powered from the PS2 bus bar.

AIRFRAME ANTI-ICING SYSTEM

DESCRIPTION

TKS R328 or TKS 80 is used for anti-icing the leading edges of the wings and horizontal stabilizer (Figure 10-4). A single WING/TAIL ANTICE time switch on the roof panel ICE PROTECTION group controls this system.

The TKS fluid is stored in a tank which holds 10.0 U.S. gallons. This quantity provides fluid for priming and protection for approximately 108 minutes. The tank filler cap is accessible from inside the cabin forward of the main entry door or behind the pilot's seat (Figure 10-5). The FULL/EMPTY indicator is located on the copilot's side console and reads FULL at 8.25 U.S. gallons and EMPTY when there is still 12 minutes' protection available.

Other airframe anti-icing components are the DC motor-driven metering pump mounted adjacent to the tank, a filter, a compensating valve, wing and tail proportioning units, and wing and tail distribution panels (Figure 10-4).

OPERATION

Before flight, the system should be primed when power is on the airplane. Set the WING/TAIL ANTICE time switch to run

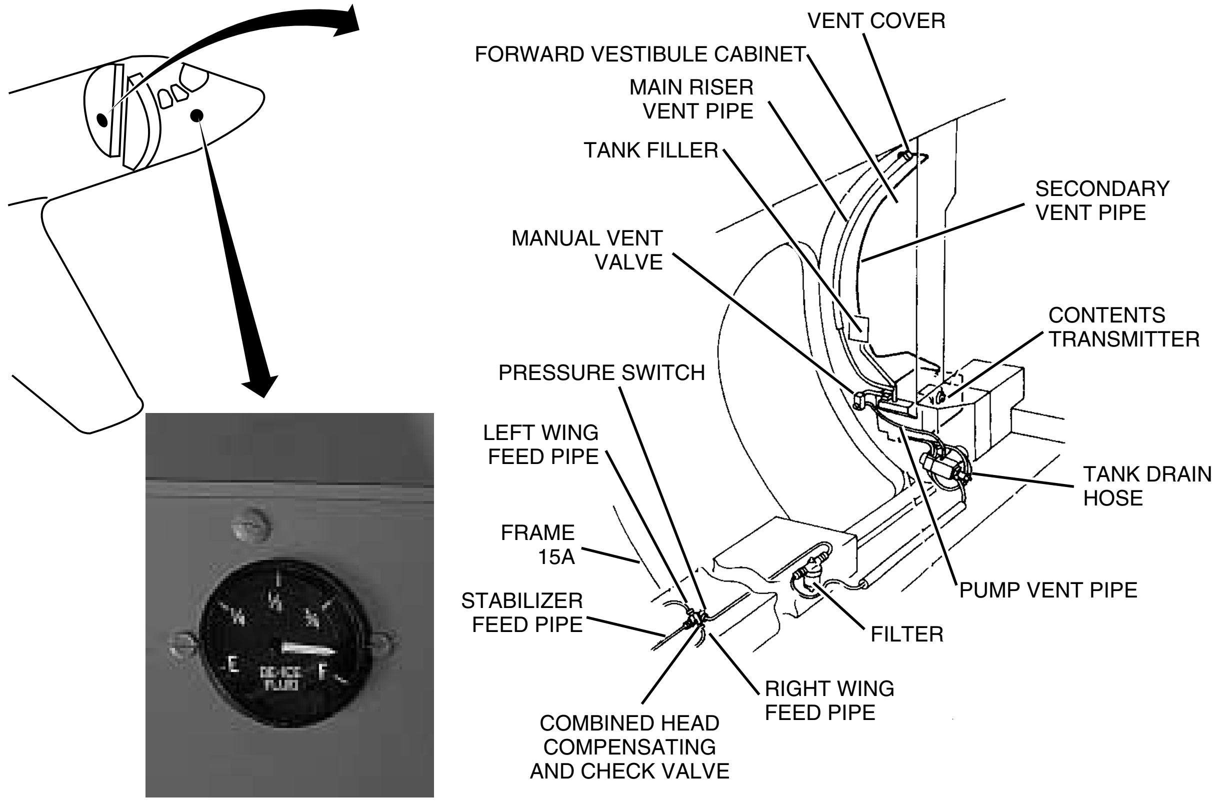


Figure 10-5. Airframe Anti-icing Fluid Tank

2 minutes. The distributor panels must then be checked for fluid flow. If it does not, reselect the time switch for brief periods until flow occurs. Refill the tank as necessary after priming.

The WING/TAIL ANTICE switch controls an electrically operated pump for up to 10 minutes. Each time it is energized, the first minute of operation is at a high flow rate, after which the system reverts to normal flow.

When the selected time expires, a warning chime is given via the audio system as the time switch returns to zero and deenergizes the pump. When this pump is operating, it feeds fluid from the tank to a filter and check valve. The valve, open during system operation, closes when the pump is inoperative and prevents undesired delivery of fluid by cabin pressure. The airframe deice pump and chime are powered from bus bar PS2.

Fluid flows from the filter to distributor panels via a compensating valve. This valve corrects system pressure levels for head variation between the wings and stabilizers. The fluid is supplied to each distribution panel via a proportional unit, which supplies the correct fluid flow to each panel cavity. From the cavity, fluid passes through a porous plastic sheet and a porous stainless steel outer skin to the atmosphere, where airflow spreads it rearward over the airfoil.

NOTE

At very low temperatures of –28°C or less, ice crystals can exist in the atmosphere, but do not present a hazard. If the airframe ice protection system is used at these low temperatures, the water/alcohol content of the fluid will evaporate, leaving solidified glycol which, together with the impinging ice crystals, can give the appearance of ice. Use of the airframe ice protection system, under these conditions, is not advisable. Therefore, operation of the WING/TAIL ANTICE time switch should be limited to the priming procedures, and additional use in flight only when weather conditions warrant.

WARNING

The aircraft must be clear of snow, ice and frost before takeoff with the exception of the following areas:

- Frost is allowable on the underside of the wings over the general area of the fuel tanks provided that the depth does not exceed 0.125 inch. If frost is present in this region, the WAT limited takeoff weight must be reduced by 1,000 pounds. The net flight path reference and fourth segment climb gradients must be obtained using a weight 1,000 pounds higher than the actual weight.

- Frost is allowable on the fuselage provided the layer is thin enough to distinguish the surface features such as paint lines or markings underneath; but, all vents, probes and ports must be clear of frost.

WARNING ANNUNCIATORS

With the TKS pump running, system low pressure is indicated by the illumination of the amber ANTICE LO PRESS annunciator on the roof panel and the amber ICE PROT ↑ repeater annunciator on the MWS panel (Annunciators Chapter).

Fluid low quantity is indicated by illumination of the amber ANTICE LO QTY annunciator on the roof panel and the amber ICE PROT ↑ repeater annunciator on the MWS panel. The ANTICE LO QTY annunciator illuminates when there is 30 minutes of TKS fluid remaining. This will show just above the 1/4 mark on the quantity gage.

All ice warning annunciators are powered from the PS2 bus bar. The fluid contents indicator power is supplied from bus bar PE when the airplane is on the ground, and bus bar PS2 when in flight. Changeover is via the squat switch.

WINDSHIELD ANTI-ICING SYSTEM

The two forward-facing windscreens and the left and right forward sidescreens are electrically anti-iced using gold film heating elements laminated between the window panes.

Power for windscreen and sidescreen heating is supplied by two 208/115 volt, frequency-wild, three-phase, engine-driven alternators. The alternators are controlled by two switches in the roof panel ICE PROTECTION group marked ALTERNATOR 1 and 2 with ON and OFF positions (see Figure 10-3). ALTERNATOR 1 powers the left windscreen, right sidescreen and left stall vane. ALTERNATOR 2 powers the right windscreen, left sidescreen and right stall vane. If an alternator fails, the other automatically supplies both windscreens, but both sidescreens are disconnected. Warning of alternator failure is indicated by amber ALTR 1 (or 2) FAIL annunciators on the roof panel and the amber ICE PROT ↑ annunciator on the MWS panel.

Windshield heating is controlled by the SCREEN HEAT L and R ON/OFF switches in the roof panel ICE PROTECTION group. The SCREEN HEAT L controls power to the left windscreen and right sidescreen. The SCREEN HEAT R controls power to the right windscreen and left sidescreen. Windshield temperature control provides two levels of temperature: a normal level and a raised level. The raised level is only utilized when airborne and an engine anti-ice switch is selected ON.

Windscreen overheat is indicated by the illumination of the amber L (or R) SCREEN OVHT and amber ICE PROT ↑ annunciators. The relevant switch should be turned OFF, then ON. If warning persists, the relevant windscreen will be controlled at a higher temperature than normal.

Sidescreen overheat is indicated by illumination of the amber SIDE SCRN OVHT and amber ICE PROT ↑ annunciators. The affected window can be identified by turning the SCREEN HEAT switches OFF and then ON one at a time. If warning persists, the relevant sidescreen will be controlled at a higher temperature than normal.

In the event of windscreen or sidescreen overheat, the associated annunciator remains illuminated, but the power supply cycles on and off as the temperature (sensed by the overheat sensor elements) falls and rises.

Outer acrylic ply failure is identified by a single crack; it will not affect the structural safety of the airplane but will result in heating failure of the panel. Switch off the appropriate SCREEN HEAT switch.

No. 1 engine alternator:

- Left windscreen heat
- Right side screen heat
- Left stall vane heat

No. 2 engine alternator:

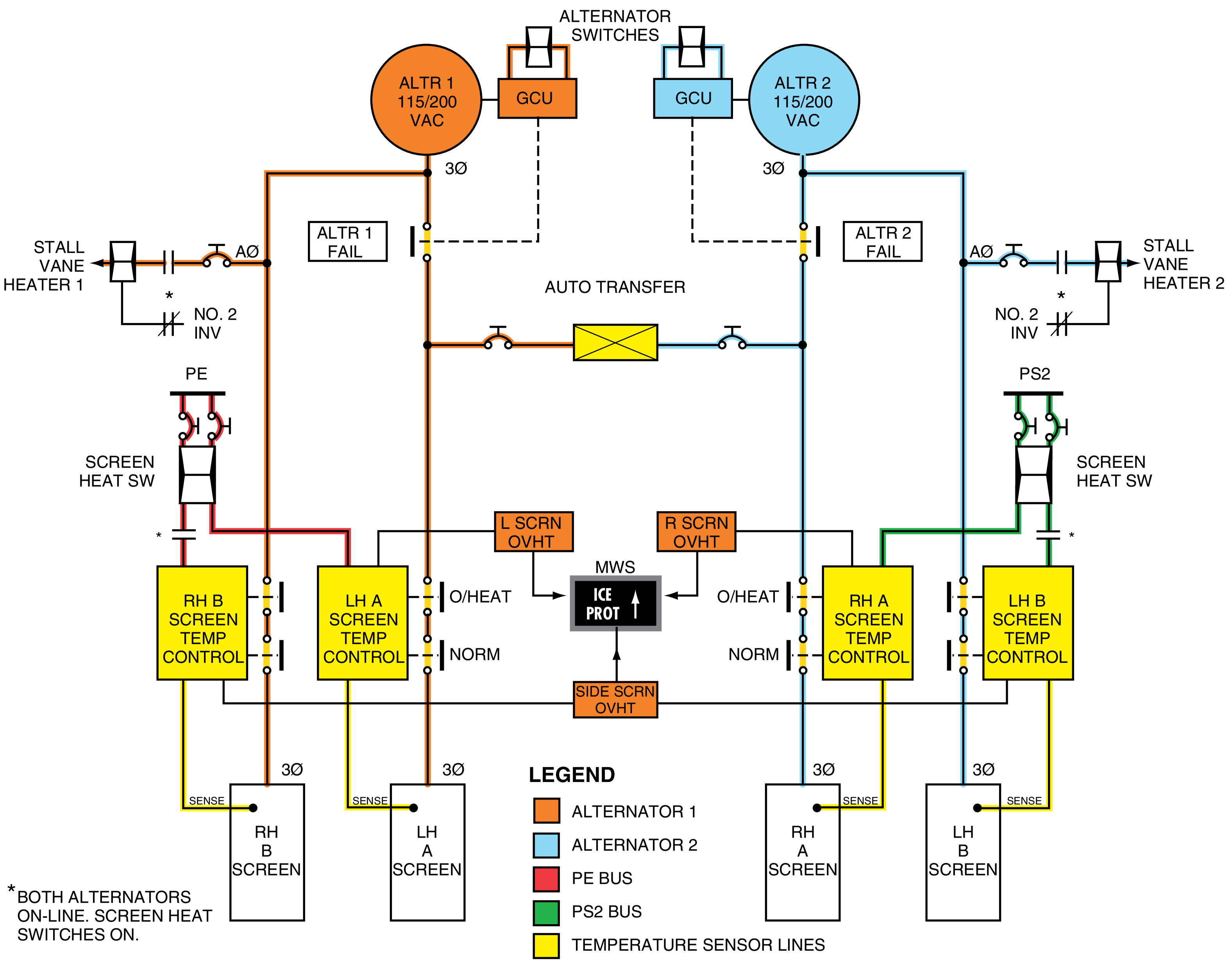
- Right windscreen heat
- Left side screen heat
- Right stall vane heat

The PE bus bar supplies power to the following:

- ALTR 1 and 2 FAIL annunciators
- L and R SCREEN OVHT annunciators
- SIDE SCRN OVHT annunciator
- Left SCREEN HEAT ON–OFF switch

The PS2 bus bar supplies power to the following:

- Right SCREEN HEAT ON–OFF switch



*BOTH ALTERNATORS ON-LINE. SCREEN HEAT SWITCHES ON.

Figure 10-6. Windshield Heat Schematic

Figure 10-6 illustrates the electrical operation of the windshield anti-icing system.

PITOT/VANE HEAT SYSTEM

The following components are electrically heated for ice protection through the PITOT/VANE HEAT L and R ON/OFF switches located in the roof panel:

- Left and right pitot heads and support masts
- Left and right forward static plates
- Two rudder bias struts
- Left and right airflow angle sensors

Each pitot head contains an electrical heating element controlled by a PITOT/VANE HEAT L or R ON–OFF switch. Each switch also controls one element of a double element heating muff fitted to each of the two rudder bias struts.

A PITOT HTR FAIL annunciator and the MWS ICE PROT ↑ repeater annunciator illuminate whenever a PITOT/VANE HEAT L or R switch is OFF, or when both switches are on and the current draw by either pitot head element is insufficient. Annunciator dimming is via the MWS dimmer.

A single ammeter and an L–R selector switch are provided (see Figure 10-3). Selecting L or R connects the ammeter to the associated pitot head and mast heater circuits. With PITOT/VANE HEAT switched on for at least 1 minute, readings of between 5 and 10 amps indicate satisfactory operation of the pitot heaters only. Actual power consumption depends on the ambient temperature.

If a PITOT HTR FAIL annunciator illuminates while the switches are on, the system ammeter is used for identification of the failed heater (current below 1 amp).

Associated instruments must be monitored for discrepancies.

WARNING

Autopilot performance may be adversely affected.

With pitot heat failure, the following equipment may fail to operate satisfactorily.

Left Pitot Heat Failed	Right Pitot Heat Failed
Air Data Computer No. 1	Air Data Computer No. 2
	Standby Airspeed Indicator
	Stall Detector Channel 3
	Additional special-order equipment, if connected

The left and right forward static plates are electrically heated. The electrical power supply to the heating element of each static plate is via the PITOT/VANE HEAT R switch through the landing gear squat switch. Heating is available only when the airplane is airborne.

Ice protection for each airflow angle sensor is provided by a vane heater element and a case heater element. The case heater element is thermostatically controlled. The left and right vane heating elements are controlled by the respective PITOT/VANE HEAT L or R switch. Vane heater failure is indicated by illumination of the L (or R) VANE HTR FAIL and ICE PROT ↑ annunciators.

Power distribution is as follows.

COMPONENTS		BUSBAR
Left pitot head heater		***PE***
Right pitot head heater		***PS2***
Pitot heater fail warning		***PE***
Rudder bias strut heaters		
	Half	***PS1***
	Half	***PS2***
Forward static plates		***PS2***

Vane heater power supply is 115 VAC and is derived as follows:

- Two windscreen alternators on line:
  - Left sensor elements from alternator No. 1
  - Right sensor elements from alternator No. 2
- One windscreen alternator off line and No. 1 and No. 2 inverters on line:
  - Elements of both sensors from No. 2 inverter
- One windscreen alternator off line and either No. 1 or No. 2 inverter off line:
  - Elements of both sensors disconnected

ENGINE ANTI-ICING SYSTEM

Air is bled from each engine to provide anti-icing for the engine nacelle inlet cowling.

An ENG ANTICE switch on the roof panel (see Figure 10-3) is provided for each engine. Each switch controls pneumatically operated bleed air valves. When the switch is in the ON position, the valves deenergize open and high-pressure bleed air is bled from the engine and directed to the inlet cowling. In addition, with ON selected and the computer switch in AUTO, electrical power is supplied to the heating elements on the $P_{T2}T_{T2}$ sensor (Figure 10-7). The ON position also controls the raised heat level for the A panel windshield in flight. In flight, the engine digital computers are reset to a schedule that incorporates a raised idle rpm to compensate for the effect on thrust.

A pressure switch in each supply line (Figure 10-8) connects electrically to an associated ENG 1 or ENG 2 A/ICE annunciator. The annunciator will come on dim initially and brighten up after two seconds if air supply is not sufficient for engine anti-icing. In addition, a white ICE PROT SELECTED annunciator on the MWS panel illuminates when either ENG ANTICE switch is on.

The engine anti-icing system is DC powered from bus bar PE.

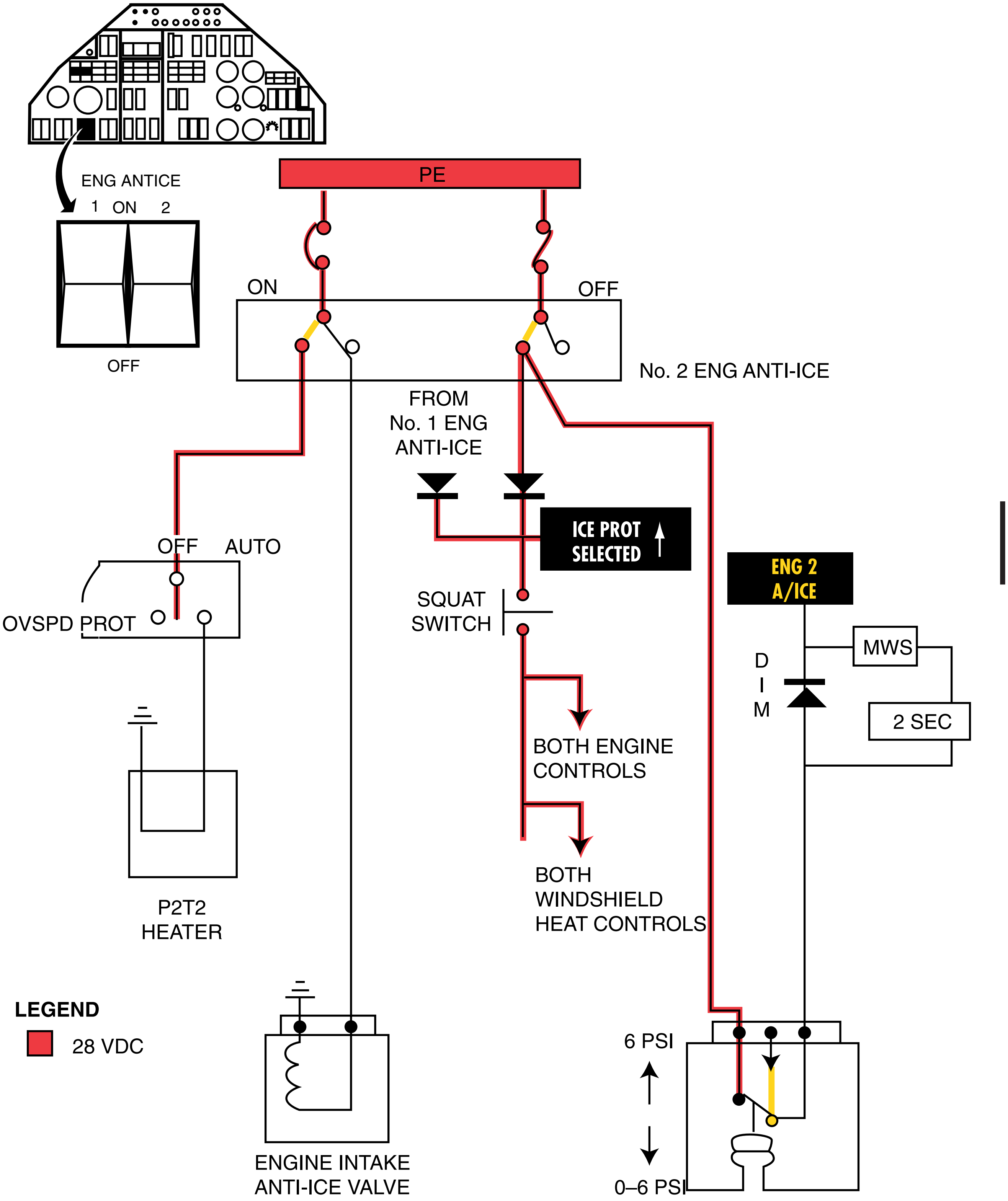


Figure 10-7. Engine Anti-ice Schematic

OPERATING PROCEDURES

The airplane is certificated for flight in icing conditions. However, the airplane must be free of snow and ice before takeoff.

Whether icing conditions are present or not, all SCREEN HEAT and PILOT/VANE HEAT switches shall always be switched on before takeoff and remain on throughout the flight.

If the airplane is cold soaked at – 10° C, windscreen heat should be on for 5 minutes prior to takeoff. If the aircraft is cold soaked at –20° C, windscreen heat should be on for 15 minutes prior to takeoff. If the times are not met, the airspeed is restricted to 257 KIAS below 8,000 feet. This restriction also applies if a windscreen becomes inoperative in flight or is turned to OFF in flight and has cold soaked at – 10° C in flight.

If the OAT or SAT is +5° C or below and is associated with conditions of high humidity (mist or fog reducing visibility to approximately 3,000 feet), or with a wet-surface runway, the ENG ANTICE and ENG IGNITION switches must be switched on. These switches should be switched off when conditions permit.

NOTE

To avoid further throttle adjustments due to slight rise in ITT, the ENG ANTICE switches should be selected to the ON position before takeoff power is set.

When takeoff is accomplished with engine anti-icing in use, ITT must be monitored during takeoff and initial climb. Allowance should be made for the adverse effects on performance when using the engine anti-icing system. If necessary, the airframe anti-icing systems may also be operated for takeoff.

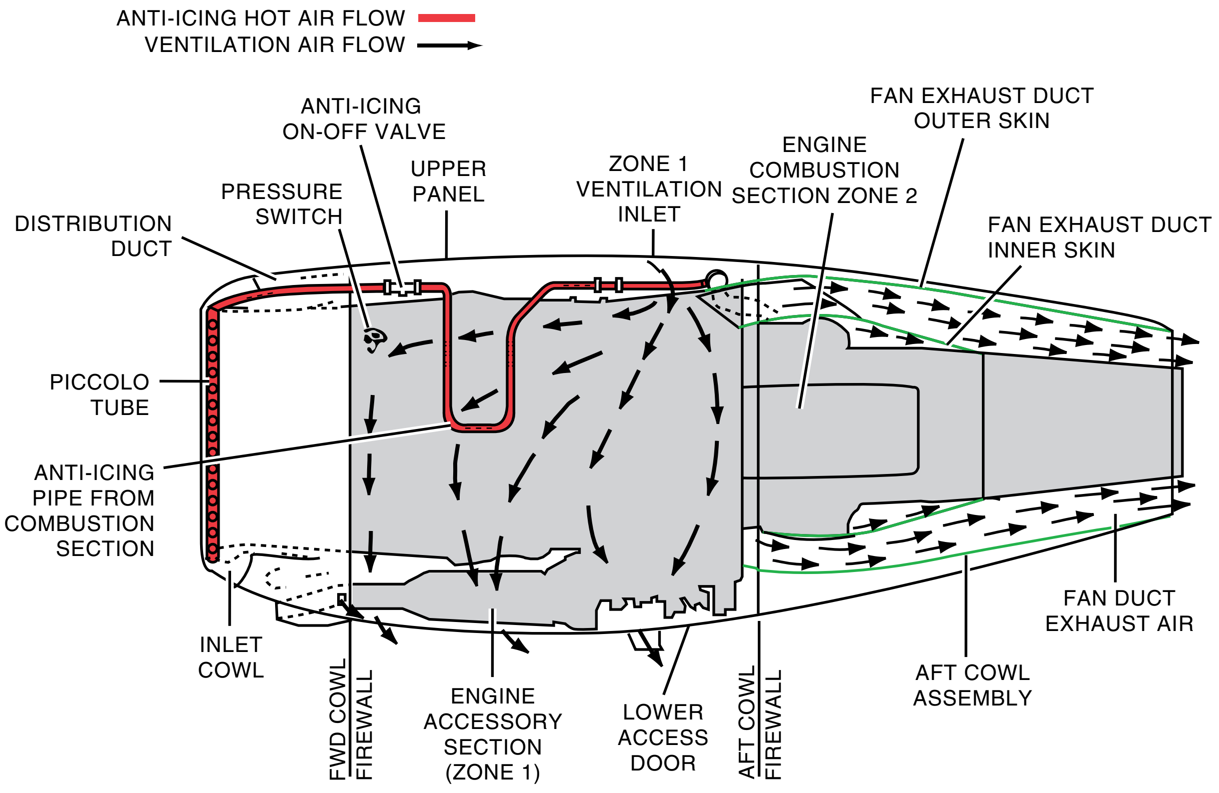


Figure 10-8. Powerplant Ventilation

To ensure that the distributors are ready for use, the airframe anti-icing system must be primed before takeoff. This is accomplished by setting the WING/TAIL ANTICE switch to run the pump for 2 minutes to prime the system regardless of weather conditions, for 30 seconds at the start of the climb, and for 2 minutes at the top of the descent.

NOTE

Normally 2 minutes' selection will prime the system. It is imperative that all panels exude fluid on the preflight check.

Do not run the airframe deicing system when the tank is empty.

If icing conditions are expected or known to be present during flight, the following actions must be accomplished:

- No. 1 ENG IGNITION switch ....ON
- No. 1 ENG ANTICE switch ....ON
- WING/TAIL ANTICE switch ....Select before entering icing and set for 10 minutes

NOTE

The airframe ice protection system should be maintained fully primed by selecting it ON for 30 seconds at the start of climb for 2 minutes at the top of descent, and if icing conditions are expected preferably for 2 minutes prior to entering icing conditions.

If icing conditions still prevail or are expected, a further period of operation should be selected prior to the time switch reaching zero. Termination of the airframe ice protection selection will be given by an audio chime.

- Airspeed ....Adjust to 230 KIAS or 0.63 indicated Mach number

NOTE

This is a recommended speed. However, if it is necessary to take advantage of the full range of airspeeds permitted for flight in icing conditions and if other conditions permit, the airspeed may be adjusted to within the limits.

- No. 2 ENG IGNITION switch........................ON
- No. 2 ENG ANTICE switch ....ON
- ENG A/ICE annunciators ....OFF

Monitor during flight in icing conditions. If an annunciator illuminates, increase engine rpm by 5%.

There is a fuel penalty with the engine anti-ice systems in use and the systems must be turned off when the aircraft is clear of icing conditions.

When slecting the ENG ANTICE switch ON, an ITT increase of 20 to 50°C can be expected and special care must be taken not to exceed the ITT limitations.

After leaving icing conditions:

- ENG IGNITION 1 and 2 switches ....OFF
- ENG ANTICE 1 and 2 switches ....OFF
- WING/TAIL ANTICE time switch ....Zero

Climb

Climb at 230 KIAS or as required in accordance with Figure 10-9, normal climb power.

Cruise

In all conditions the aircraft has sufficient performance to be able to cruise above 30,000 feet, where icing is unlikely to occur.

If it is necessary to cruise in an icing layer, the long-range speed should be used. It is usually more economical to cruise below the icing layer rather than in it.

Holding

Holding should be done at the normal speed.

NOTE

The procedural use of 15° flaps for holding is not permitted in icing conditions.

Descent

When descending into icing conditions, select the airframe ice protection system on for 2 minutes before entering icing (approximately 5,000 feet above the cloud).

With ENG ANTICE selected in flight and the thrust lever at idle, a raised N_2 is automatically supplied at which adequate intake and engine anti-ice are available.

NOTE

The procedural use of 15° flaps for descent is not permitted in icing conditions.

Descent should be made at 230 KIAS or as required in accordance with Figure 10-9. Thrust levers may be closed. Some airbrakes may give a rate of descent of about 3,000 feet per minute. Higher IAS, up to the maximum, may be used if required to give a higher rate of descent.

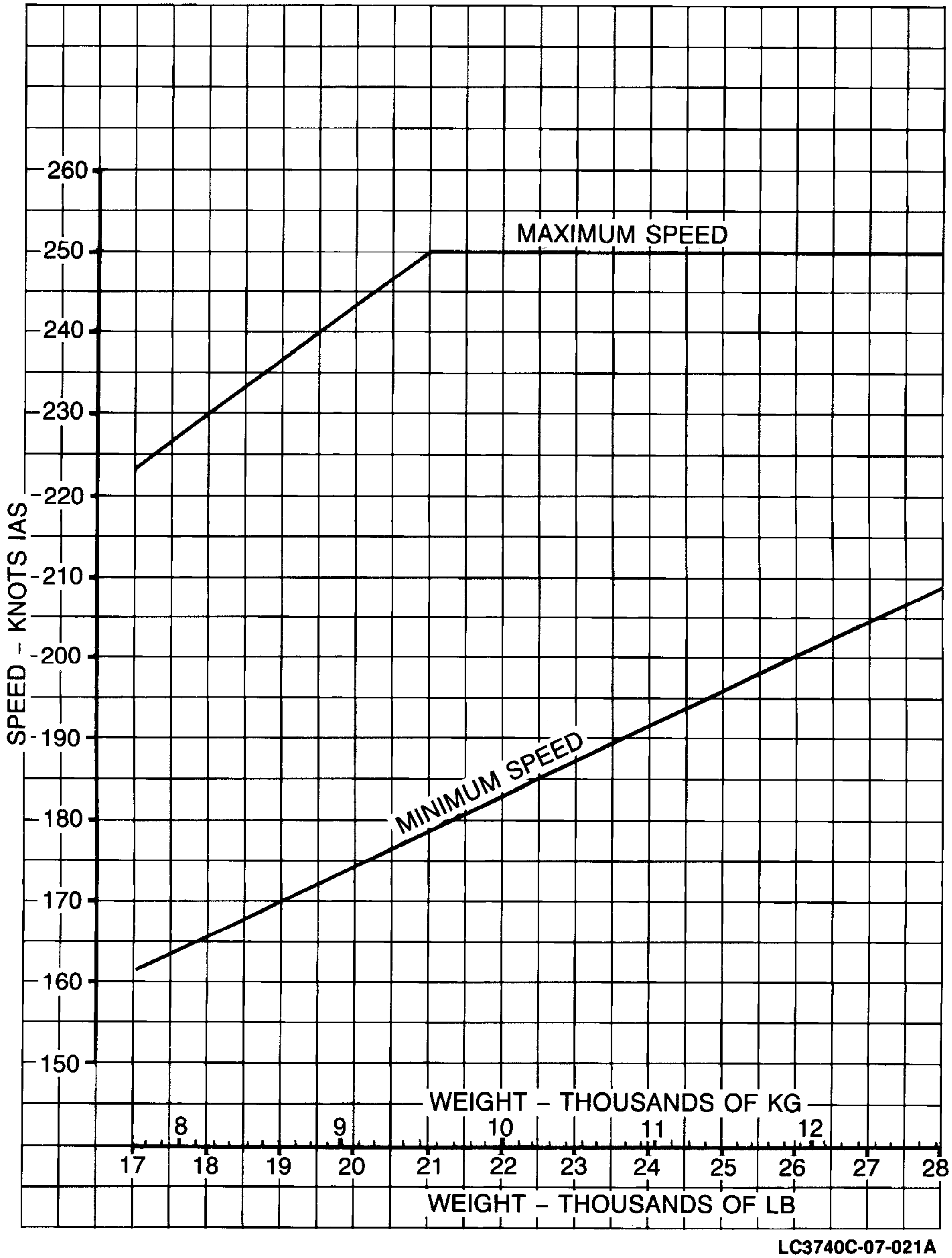


Figure 10-9. Speed for Use in Icing Conditions

QUESTIONS

1. The following component(s) is deiced by use of hot bleed air:
A. Engine inlet cowlings
B. P_2T_2 sensors
C. Rudder bias struts
D. Horizontal stabilizer

2. The following component(s) or system(s) is controlled through the left main gear squat switch:
A. Pitot heat and airflow angle sensor heat
B. Ice detector and airflow angle sensor heat
C. Airframe deicing system pump
D. Windshield anti-icing system

3. For takeoff or landing approach in icing conditions, the following switches must be turned on:
A. ENG IGNITION, ENG ANTICE, WING/TAIL ANTICE
B. ENG ANTICE, WING/TAIL ANTICE, ICE DET
C. ENG ANTICE, WING/TAIL ANTICE, demist control
D. ICE DET, ENG IGNITION, ENG ANTICE

4. The following switches provide DC power to the rudder bias heater muffs:
A. ALTERNATORS 1 and 2
B. PITOT/VANE HEAT
C. SCREEN HEAT
D. ENG IGNITION

5. The following statement(s) is true:
A. The engine inlet cowling is heated by hot bleed air.
B. With the ENG ANTICE switch energized and the computer switch in AUTO, electrical power is supplied to heating elements on the P_2T_2 sensor.
C. When not in icing conditions, engine anti-icing must not be used for more than 10 seconds in ambient air temperatures above 10° C.
D. All the above

6. The ICE PROT ↑ annunciator on the MWS panel does not illuminate when:
A. The ice detector detects ice.
B. Low pressure is detected in the airframe deicing system.
C. TKS fluid quantity is low.
D. PITOT/VANE HEAT is on and operating normally.

7. The following component may fail to operate satisfactorily when the left pitot heat is inoperative:
A. ADC No. 1
B. ADC No. 2
C. Standby airspeed indicator
D. Standby altimeter

8. The ENG 1 (or 2) A/ICE annunciator (when illuminated) indicates:
A. That the respective engine anti-icing is on and operating normally
B. That anti-icing air pressure is available at sufficient pressure
C. That anti-icing air pressure is not available at sufficient pressure
D. That ice has been detected on the P_2T_2 sensor

9. The PITOT AMPS ammeter checks amperage of:

A. The left or right pitot heating elements (as selected)
B. Both pitot heating elements simultaneously
C. The left or right vane heating elements (as selected)
D. The left or right pitot and vane heating elements (as selected)

10. The forward static plates are heated:

A. On the ground when OVRD is selected
B. At low heat only when on the ground
C. Anytime the PITOT/VANE HEAT R switch is on
D. Only when airborne

11. When engine anti-icing is required, the ENG ANTICE switches should be turned on:

A. When established in initial climb
B. After takeoff power is set
C. Before takeoff power is set
D. A minimum of 5 minutes before takeoff

12. The TKS fluid supply lasts for approximately:

A. 30 minutes
B. 45 minutes
C. 61 minutes
D. 108 minutes

13. The airframe deicing system:

A. Should be primed before flight
B. Should be primed only before flight into known icing conditions
C. Will deice satisfactorily while on the ground
D. Should never be activated until airborne

14. The following ICE PROTECTION switches shall always be switched on before takeoff and remain on throughout the flight:

A. SCREEN HEAT and ENG ANTICE
B. ENG IGNITION and PITOT/VANE HEAT
C. ENG ANTICE and ENG IGNITION
D. PITOT/VANE HEAT and SCREEN HEAT

CHAPTER 11
AIR CONDITIONING

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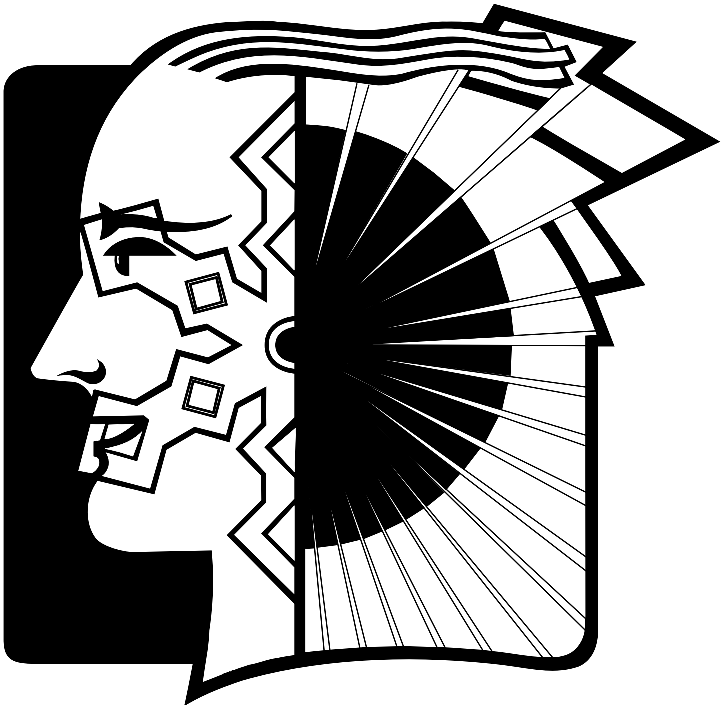
CHAPTER 11
AIR CONDITIONING



INTRODUCTION

The Hawker 800 XP airplane incorporates an air-conditioning system which includes a cooling system, an air distribution system, controls, and indicators. The air-conditioning system provides a comfortable environment for the crew and passengers. On the ground, the system's air source is either the auxiliary power unit (APU) or the engines; in flight, the air source is the engine pneumatic system. The conditioned air is used to control pressurization of the airplane, which is discussed in Chapter 12, "Pressurization."

GENERAL

The cooling system includes a refrigeration unit, a cabin temperature control valve, a low-limit temperature control valve, and a water separator. The resulting temperatures are determined by the positions of the valves.

Bleed air is tapped from both engines and conditioned to acceptable temperature levels by an air conditioning pack incorporating a three-wheel air cycle machine.

Conditioned air is ducted to the flight compartment and passenger cabin to provide ventilation, heating and pressurization. Temperature levels, set manually, are automatically controlled.

On the ground, conditioned air is supplied by the APU and the air cycle machine when the main engines are not running.

PRESSURE REGULATOR AND SHUTOFF VALVE (PRSOV)

From each main air valve (MAV), the air is ducted via a nonreturn valve (NRV), and then via a common duct to a pressure regulator and shutoff valve (PRSOV) unit and a pressure reducing and flow restricting venturi to an air cycle machine (ACM).

AIR CYCLE MACHINE (ACM)

The air cycle machine (ACM) consists of a three-wheel cold air unit (CAU), and primary and secondary heat exchangers which are cooled by ram air. Excess water is removed by a water separator at the exit from the ACM. To enhance the cooling performance, this water is ejected back into the ram-air flow above the primary heat exchanger (Figure 11-1).

OVERSPEED PROTECTION

A pressure switch, downstream of the PRSOV, protects the ACM from overspeed should the PRSOV fail. The pressure switch operates on a rising pressure of 40 psi to signal the No. 2 MAV to close.

The No. 1 MAV remains open and the maximum flow to the CAU is restricted by the venturi in the No. 1 LP bleed duct.

Closure of the No. 2 MAV will be indicated by the MWS panel annunciator, MAIN AIR VLV 2, illuminating.

NOTE

The illuminated MAIN AIR VLV 2 annunciator indicates that the position of the No. 2 MAV does not agree with the selection made on the MAIN AIR VLV 2 switch.

This annunciation is not unique to the overspeed condition. However, the overpressure condition also triggers a second indication—a magnetically latched doll's eye indicator in the rear equipment bay. This indication remains triggered until reset by ground-servicing personnel.

The closure of No. 2 MAV results in a decrease in system pressure. To avoid cycling of the valve, a latch circuit maintains the closed signal. The system can be reset by selecting the MAIN AIR VLV 2 switch to CLOSE. This action causes the MAIN AIR VLV 2 annunciator to go out. If the system pressure has fallen, the No. 2 MAV will subsequently open when the switch is selected OPEN.

COLD AIR UNIT (CAU)

The CAU comprises a fan, a compressor and a turbine mounted on a common shaft.

Bleed air from the venturi enters the ACM via the primary heat exchanger, which cools this air to an acceptable level for the CAU. The primary exchanger is cooled by ram air from an intake at the base of the fin.

The cooled air from the primary heat exchanger enters the CAU compressor, which raises its pressure and temperature before it is cooled by the secondary heat exchanger. The secondary heat exchanger uses ram air as the cooling medium.

From the secondary heat exchanger, the air is expanded and cooled through the turbine. The energy extracted in this process is used to drive the compressor and the fan. The fan is used to draw cooling air through the ram-air system.

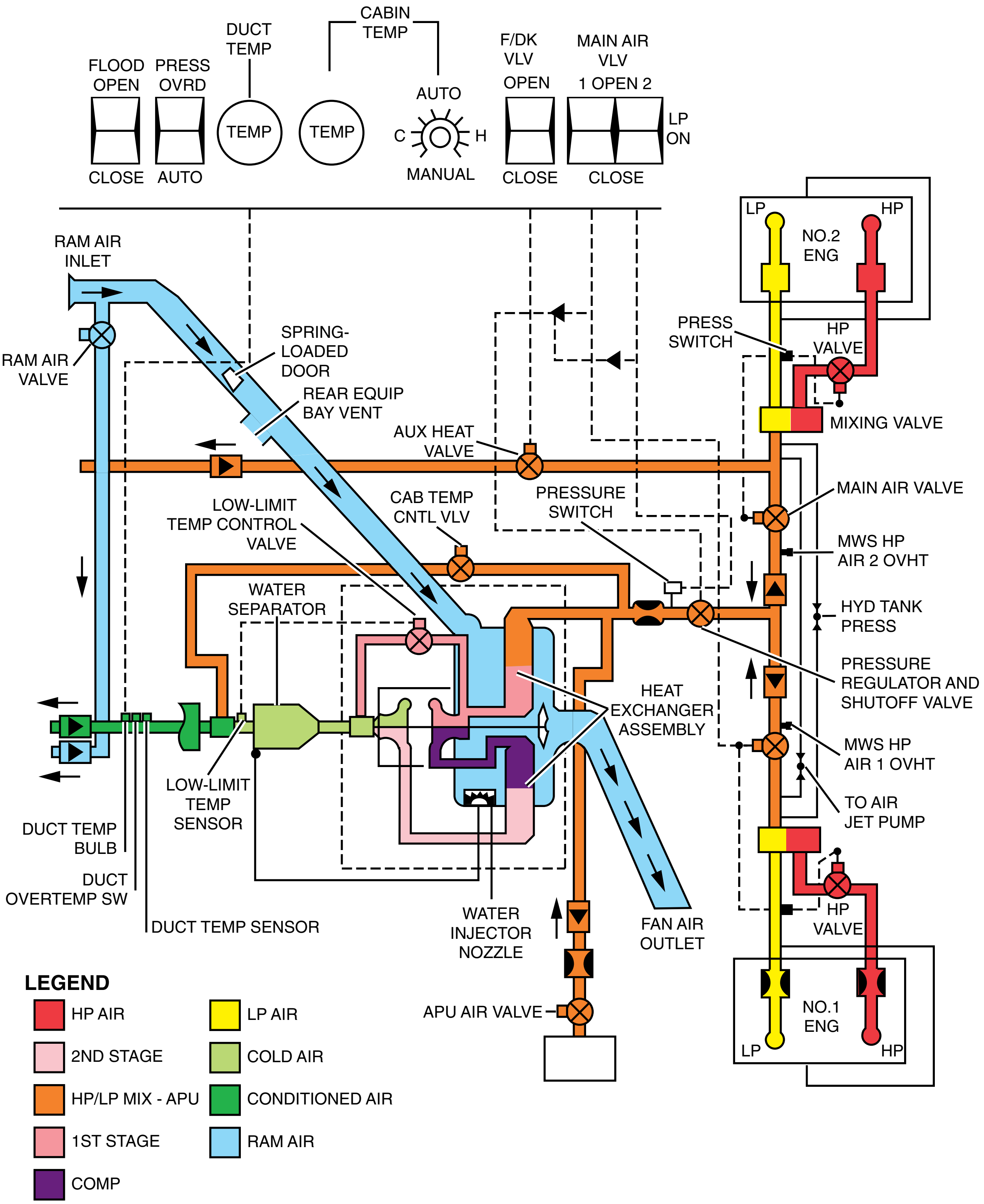


Figure 11-1. Air Cycle Machine (ACM)

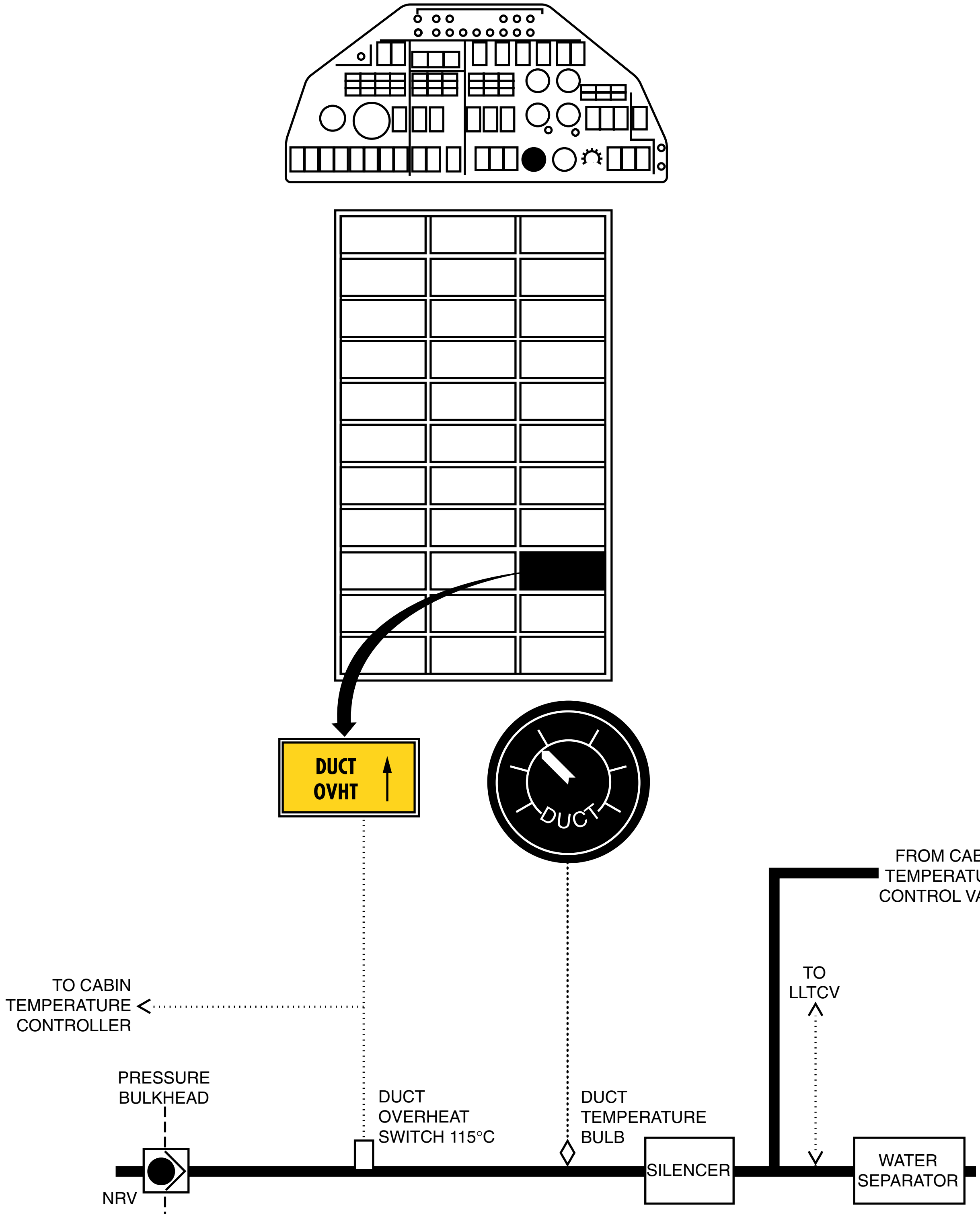

DUCT OVHT
DUCT
FROM CABIN TEMPERATURE CONTROL VALVE
TO CABIN TEMPERATURE CONTROLLER
TO LLTCV
PRESSURE BULKHEAD
DUCT OVERHEAT SWITCH 115°C
DUCT TEMPERATURE BULB
SILENCER
WATER SEPARATOR
NRV

Figure 11-2. Duct Temperature Indication and Overheat Circuits

At the exit from the turbine, the air temperature is below dewpoint and water is condensed out in the form of fog. To prevent the formation of ice, the temperature of the air before entry into the water separator is controlled by the low-limit temperature control system (see below).

The water separator extracts approximately two-thirds of the water from the airflow. The drained water is injected back into the heat exchanger cooling air to enhance the cooling capacity. The temperature of the air downstream of the water separator is controlled by the cabin temperature control valve.

LOW-LIMIT TEMPERATURE CONTROL SYSTEM (LLTCV)

The air temperature at the inlet to the water separator is limited by a control circuit to a minimum temperature of about 2° C to prevent freezing and consequent blockage of the water separator (Figure 11-1). Warm air is routed from the primary heat exchanger outlet via the low-limit temperature control valve (LLTCV) to the inlet of the water separator.

The operation of the LLTCV is controlled by a low-limit control sensor in the duct downstream of the water separator. The low-limit temperature control system operates independently of any other system and can override cold selections made by the flight crew.

DUCT TEMPERATURE

A DUCT TEMP indicator, on the roof panel, is connected to a temperature sensor located in the duct downstream of the silencer (Figure 11-2). This temperature may be considered as the cabin air inlet temperature.

A duct overtemperature sensor, set at 116° C, will cause a DUCT OVHT amber annunciator to be illuminated on the MWS, should the temperature be exceeded. The overtemperature signal will also cause the cabin temperature control valve to be motored fully closed. When the temperature in the duct falls, normal automatic control is restored and the DUCT OVHT annunciator is extinguished.

CABIN TEMPERATURE CONTROL SYSTEM

Temperature of the air delivered to the cabin may be varied by the crew via selections on the CABIN TEMP AUTO–MANUAL/COOL–HOT temperature selector. There are two modes of operation for the cabin temperature system: AUTO or MANUAL (Figure 11-3).

AUTO MODE

In the AUTO mode, the required temperature in the cabin is selected by positioning the selector between COOL and HOT. Temperature is not indicated on the temperature selector; however, a full COOL selection corresponds to 18.32° C (65° F), and a full HOT selection corresponds to 31.1° C (88° F) cabin temperature.

A temperature controller using signals from the selector switch, a duct temperature sensor, and a cabin temperature sensor unit, determines whether an increase or decrease of temperature is required. Resulting signals from the controller cause the cabin temperature control valve to open or close accordingly.

A CABIN TEMP indicator, on the flight compartment roof panel, is connected to a temperature bulb at the right forward cabin bulkhead position. An electrically operated fan draws air through a grill in the cabin and across the temperature bulb and the cabin temperature sensor.

MANUAL MODE

The MANUAL mode is selected by moving the CABIN TEMP selector through a detent to the MANUAL spring-loaded center-off position.

Holding the selector to either COOL or HOT (as required) directly controls the position of the cabin temperature control valve. Releasing the CABIN TEMP selector to the center-off position stops the cabin temperature control valve in the attained position.

The duct overtemperature limiting sensor and the low-limit control system remain operative in the MANUAL mode.

AIR DISTRIBUTION SYSTEM

The air distribution system (Figure 11-4) is a network of ducting and outlets used to route conditioned air to the cabin and the cockpit.

The flight compartment air duct is branched to supply conditioned air to louvres in both side consoles.

Normally, the only air source for the air distribution system is temperature-controlled air from the air-conditioning system. If additional heating is required, bleed air from the No. 2 engine can be routed directly to the flight compartment when the auxiliary heating valve is opened.

CABIN FLOOD SWITCH

Temperature-controlled air enters the pressure cabin through a silencer, a nonreturn valve and an electrically operated valve. This is controlled by a CABIN FLOOD OPEN–STOP–CLOSE switch on the flight compartment roof panel.

The switch is spring-loaded to the center (STOP) position. Holding the switch to one or the other positions allows the valve to move to the selected position.

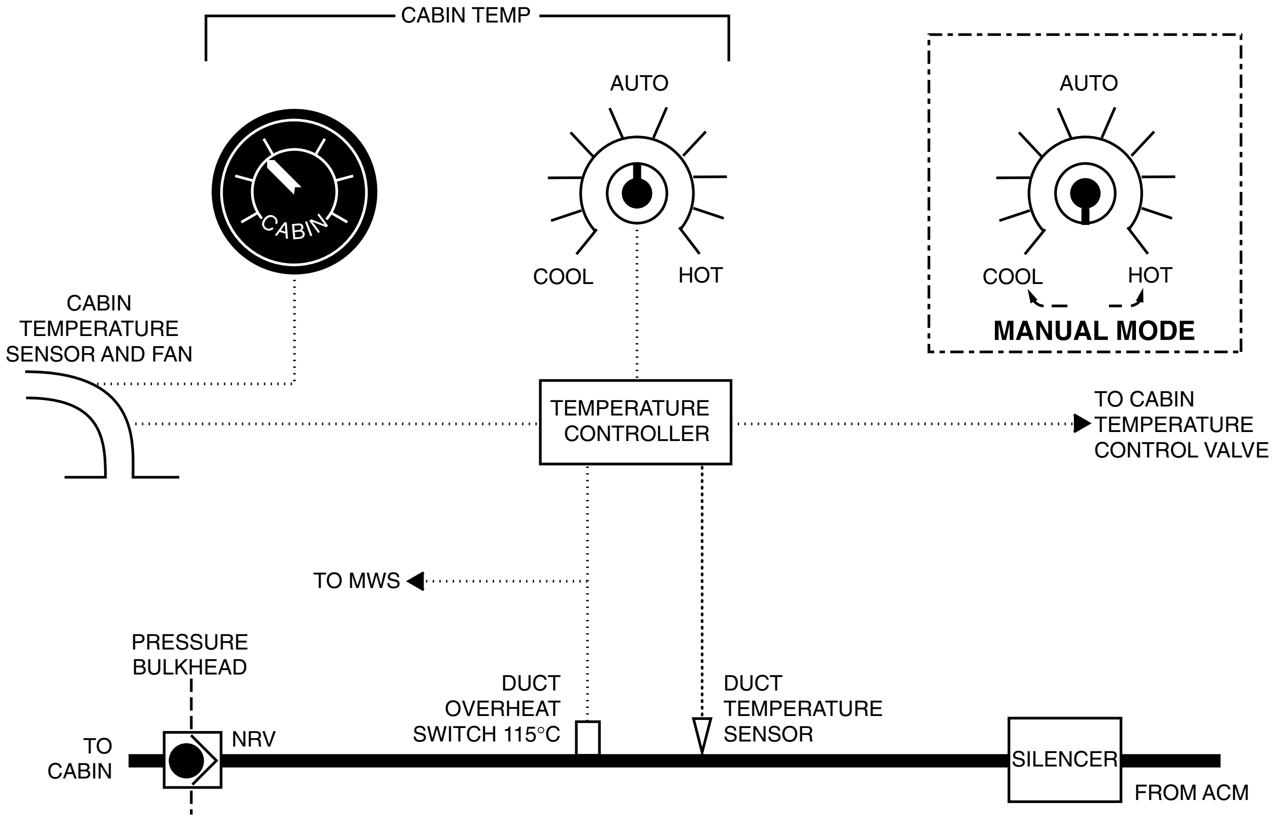


Figure 11-3. Cabin Temperature Control

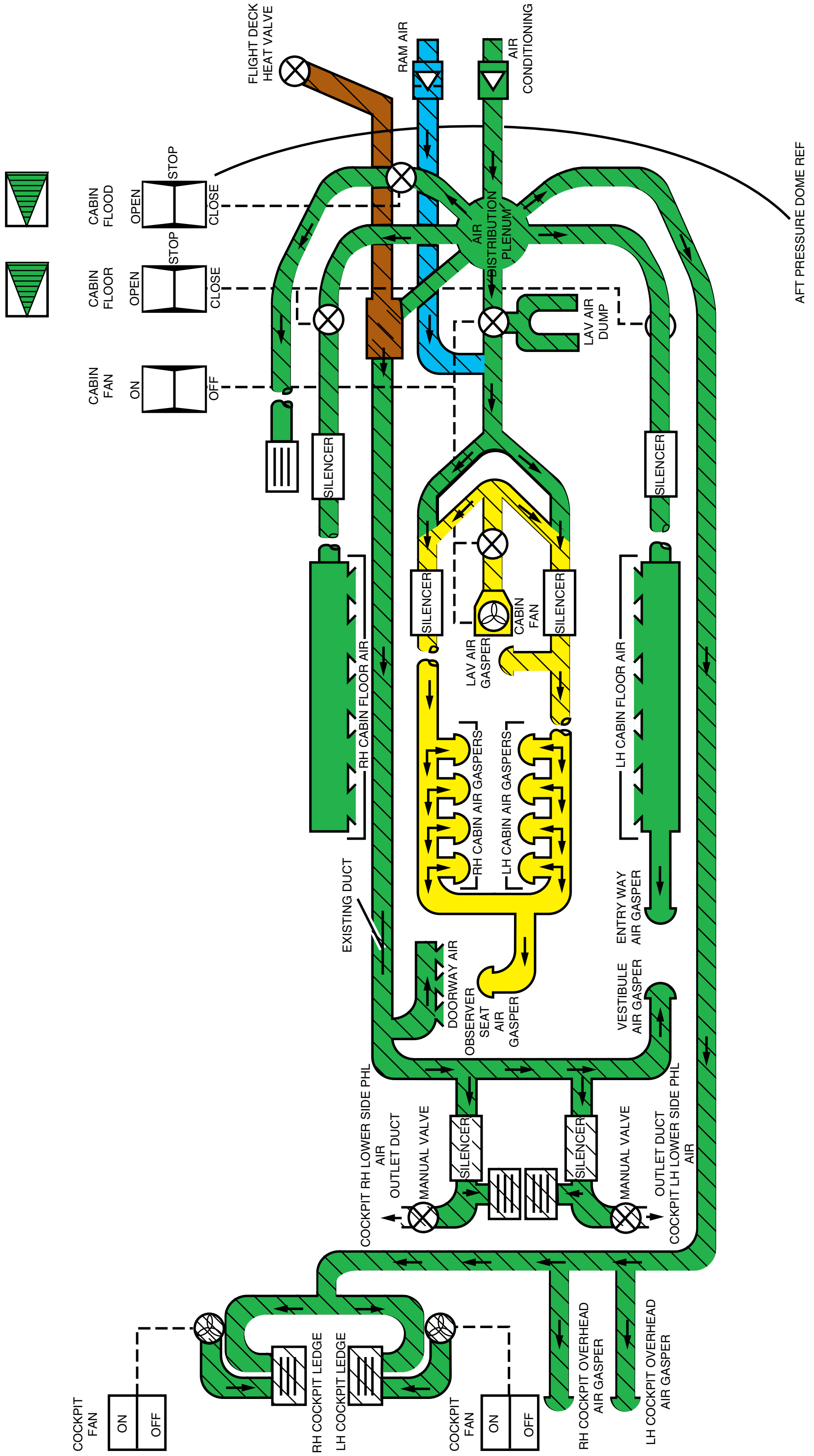


Figure 11-4. Air Distribution Block Diagram

With the CABIN FLOOD selected OPEN, all the conditioned air is supplied to a cabin flood outlet, at roof level at the right rear of the passenger cabin.

With the CABIN FLOOD selected CLOSE, the air supply to the cabin flood flow outlet is cut off.

A green LED indicator above the switch displays the valve position.

CABIN FAN SWITCH

An electrically operated fan draws air from the rear of the cabin and recirculates it via individual controllable outlets on each passenger service unit (one outlet per passenger seat position). The recirculated air is also supplied to two further outlets in the flight compartment.

The fan is controlled from a FAN ON–OFF switch in the ENVIRONMENTAL section of the flight compartment roof panel.

CABIN FLOOR SWITCH

Temperature-controlled air enters the pressure cabin through a silencer, a nonreturn valve and two electrically operated valves. This is controlled by a CABIN FLOOR OPEN–STOP–CLOSE switch on the flight compartment roof panel.

The switch is spring-loaded to the center (STOP) position. Holding the switch to one or the other positions allows the valve to move (or inched) to the selected position.

With the CABIN FLOOR selected OPEN, all the conditioned air is supplied to both cabin floor ducts.

With the CABIN FLOOR selected CLOSE, the air supply to both cabin floor ducts is cut off.

A green LED indicator above the switch displays the valve position.

GENERAL SYSTEM CONTROL

In cold weather operation, the floor ducts should be opened to conditioned air and the cabin fan turned on to recirculate the cabin air through the cabin air gaspers for balanced cabin heating.

In hot weather operation, the cabin floor ducts may be positioned as required for balanced cooling of the cabin.

After the aircraft reaches altitude and cold soaks, the floor ducts should be opened as needed, to maintain the desired cabin temperature. The cabin fan can be turned on, if necessary, to recirculate the cabin air through the cabin air gaspers for a more balanced cabin temperature.

NOTE

> It is recommended that when the cabin fan is in operation, the floor valves should not be closed more than halfway for efficient operation.

AUXILIARY HEATING SYSTEM

The auxiliary heating system supplies the flight compartment with a variable supply of hot air via a heat augmentor in the main supply system. Control of the auxiliary system valve is by the F/DK VLV OPEN–CLOSE switch on the roof panel. The supply is tapped from upstream of the main air valve of the No. 2 engine via a nonreturn valve.

If, while the flight compartment auxiliary heating valve is open, an overheat of 115°C is detected in the outlet duct, then the flight compartment auxiliary heating valve is closed. It remains closed until selected manually open.

RAM AIR

Ram air from the dorsal air intake is used primarily as a cooling medium for the heat exchanger, but it can also be introduced into an unpressurized cabin for ventilation purposes. The cabin supply is ducted from the ram-air intake and fed into the cabin through an electrically operated ram-air valve and a nonreturn valve, and into the flood flow duct. The ram-air valve is controlled by a microswitch which operates when the DUMP VALVE lever is selected fully OPEN.

The ram air passes through the CAU heat exchanger assisted by a fan driven by the CAU turbine, and is ducted to atmosphere through a grille in the aircraft skin.

If the intake becomes obstructed, a spring-loaded door in the intake duct opens inwards to permit air from the rear equipment bay to enter the intake and supply the CAU heat exchanger.

REAR EQUIPMENT BAY VENTILATION

A tapping is taken from the dorsal fin air intake to provide an air supply for rear equipment bay ventilation. There are no controls or indications for this supply.

GROUND AIR CONDITIONING

Ground air conditioning can be supplied by the auxiliary power unit (APU). The APU air supply is fed through a shutoff valve and a nonreturn valve into the main supply duct downstream of the pressure regulator and shutoff valve (PRSOV). APU air is controlled by the APU AIR VLV OPEN–CLOSE switch on the flight compartment roof panel.

ELECTRICAL POWER SOURCES

PE bus bar

- Flight deck heat valve
- HP air overheat warning
- HP bleed on–off valve
- Main air valves 1 and 2
- Cabin temperature (manual)
- Duct overheat detect

PS1 bus bar

- temperature indicator

PS2 bus bar

- Cabin temperature indicator
- Cabin temperature (auto)
- Cabin temperature sensor

LIMITATIONS

There are no limitations imposed on the air-conditioning system.

QUESTIONS

1. The main air valves and flight deck heat valve must be ________ prior to takeoff or landing.
   - A. Full open
   - B. LP on
   - C. Closed
   - D. A or B

2. Which component in the air conditioning system determines the temperature of the air prior to entering the water separator?
   - A. Cooling turbine
   - B. Cabin air sensor
   - C. Heat exchanger
   - D. Low limit temperature

3. When the duct temperature limiter senses a duct temperature of 116°:
   - A. The limiter turns on the DUCT OVHT annunciator only.
   - B. The limiter turns on the DUCT OVHT annunciator and automatically gives partial reduction in duct temperature.
   - C. The limiter automatically reduces duct temperature only.
   - D. The limiter automatically shuts down the flow control units.

4. Cabin delivery air temperature is controlled by operation of the:
   - A. No. 1 main air valve
   - B. No. 2 main air valve
   - C. Cabin temperature control valve
   - D. AUX heat valve

5. A pressure switch downstream of the PRSOV, protects the ACM from overspeed if pressure rises above?
   - A. 27 psi
   - B. 40 psi
   - C. 60 psi
   - D. 57 psi

6. The switch that controls the auxiliary heating valve to provide additional heating to the cockpit is the:
   - A. AUX HEAT switch
   - B. EMER PRESS switch
   - C. PRESSN GEAR OVRD–AUTO switch
   - D. F/DK VLV switch

7. The approximate time the F/DK VLV switch must be held in the OPEN position in order for the heat valve to fully open is:
   - A. 5 seconds
   - B. No time limit
   - C. 20 seconds
   - D. 10 seconds

CHAPTER 12
PRESSURIZATION

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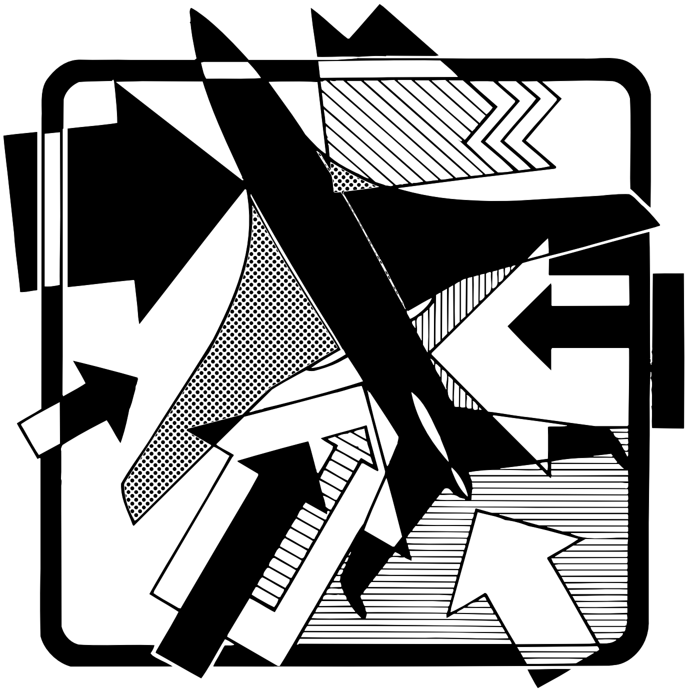
PRESSURIZATION



INTRODUCTION

The Hawker 800 XP airplane incorporates a pressurization system that includes a controller, two outflow/safety valves, and a venturi. Cabin pressure is automatically controlled by the two outflow/safety valves which are pneumatically operated to maintain the 8.55 psi differential between cabin and ambient pressure. These valves govern the exhausting of cabin air to the atmosphere. Safety devices completely shut off the outflow when the cabin altitude reaches a preset value.

The outflow/safety valves also provide inward and outward relief for both negative and positive differential conditions in order to protect the airplane's structure. A full range of manual control is available in the event of a malfunction of the automatic control by utilizing the manual pressurization controls on the copilot's side console. A dump valve provides cabin pressure control when the normal system control is inoperative. It can also be used in conjunction with a ram-air supply to ventilate the cabin while it is unpressurized.

The purpose of the pressurization system is to ensure crew and passenger survival and comfort regardless of the airplane's altitude.

GENERAL

When pressurization and air conditioning are in use, a supply of air is fed into the pressure cabin. Outflow is controlled by a cabin pressure controller and by two outflow/safety valves. The pressure controller regulates both the discharge of the air and the discharge rate. The rate control affects the cabin rate of climb/descent.

The pressurization system can maintain a cabin altitude of 8,000 feet at an airplane altitude of 43,000 feet at 8.55 psi maximum cabin differential pressure.

If a crewmember selects a cabin differential greater than 8.55 psi, a safety device on the outflow/safety valve prevents the cabin pressure from exceeding 8.6 to 8.8 psi. If an attempt is made to position the airplane so that the 8.55 psi cabin differential pressure is exceeded, the same safety device causes the cabin pressure to decrease at the same rate the airplane climbs.

During a rapid descent, if the ambient pressure exceeds cabin pressure, another safety device allows ambient pressure to enter the cabin, preventing a negative cabin differential pressure from exceeding 0.5 psi.

The 8,000 feet cabin altitude ensures that no passenger discomfort associated with hypoxia is experienced during normal high-altitude flight.

DESCRIPTION

The pressurization system (Figure 12-l) includes a CABIN CONTROLLER, two outflow safety valves, two pneumatic relays, a venturi, an air jet pump, two pressure regulators, a fan-operated venturi, and controls and indicators. Two controls labeled “MANUAL CABIN ALTITUDE CONTROL” and “PRESSURIZATION CONTROL” permit manual control of the system.

CABIN CONTROLLER

The CABIN CONTROLLER (pressure controller) (Figure 12-1), located on the copilot’s instrument panel, ensures that the outflow/safety valves maintain a selected cabin altitude and that the cabin air pressure changes at an acceptable rate during climb or descent. The controller has two knobs. The center knob is labeled “CABIN”; the knob on the lower left is labeled “RATE.”

The CABIN knob controls rotation of a dial containing an inner and outer scale. Both scales indicate altitude and are graduated in feet x 1,000. The outer scale displays the selected cabin altitude. With the cabin altitude selected, the airplane’s altitude at which the maximum cabin differential will be reached is displayed on the inner scale. This provides an 8,000 feet cabin altitude at an actual airplane altitude of 43,000 feet.

The RATE knob is used to adjust cabin altitude rate of change. This range is from approximately 2,000 fpm (MAX setting) to 50 fpm (MIN setting). Positioning the arrow vertically provides a rate of change of approximately 500 fpm.

OUTFLOW/SAFETY VALVES

Controlled by the pressure controller, the two outflow/safety valves permit a controlled discharge of cabin air. The valves incorporate safety devices which prevent a pressure buildup in excess of the maximum cabin differential of 8.6 to 8.8 psi and of the negative pressure differential of 0.5 psi. They are located on the rear pressure bulkhead.

PNEUMATIC RELAYS

Two identical pneumatic relays sense pressure changes within the controller and transmit these changes to the outflow/safety valves. The relays are located on the rear pressure bulkhead. They amplify the pressure changes from the controller located in the cockpit, thus improving system response.

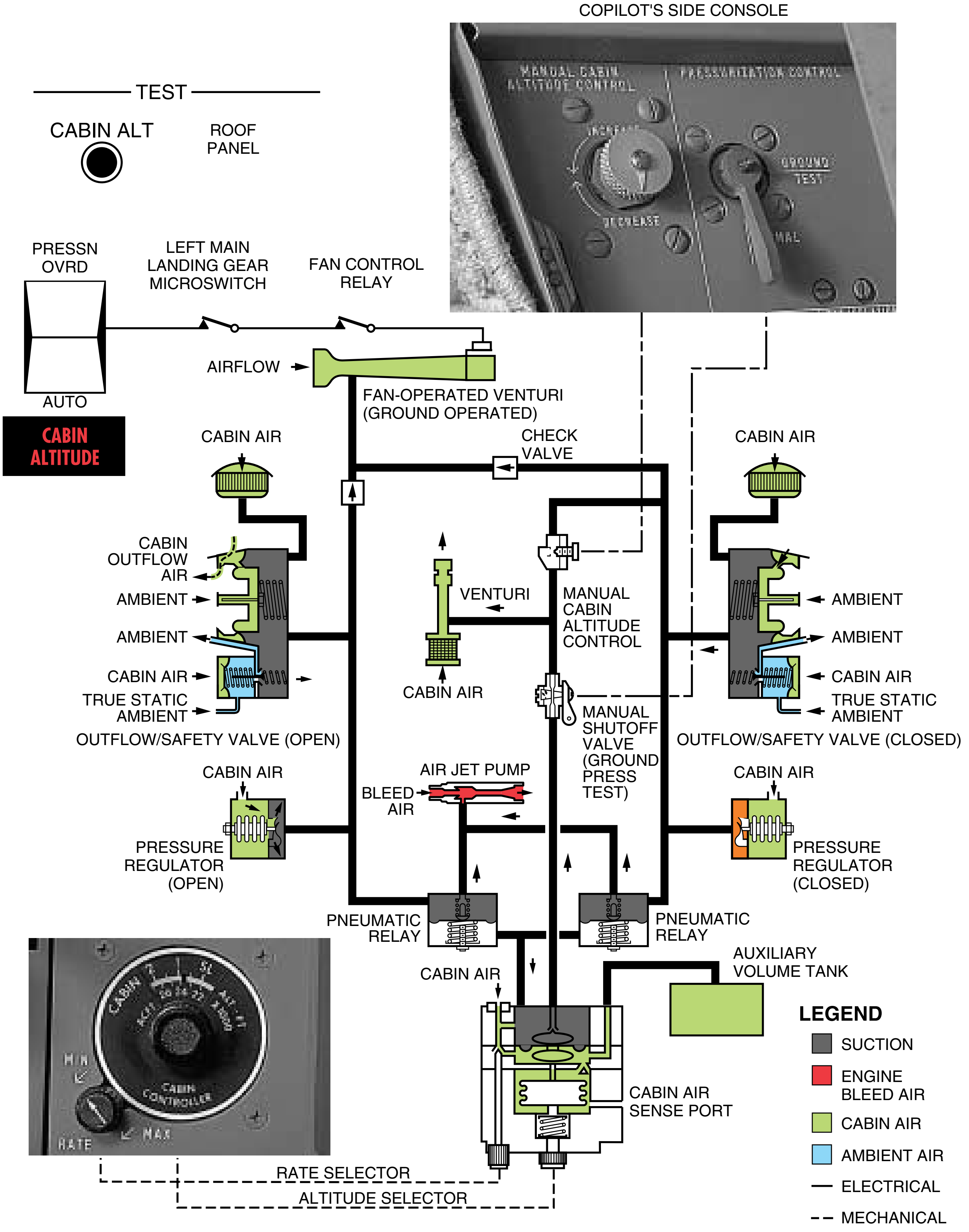


Figure 12-1. Pressurization System Diagram

VENTURI

A venturi located in the forward luggage bay area extracts air from the autocontroller during normal operation. In manual operation, it extracts air from one of the outflow/safety valves via the manual control valve on the copilot's side console.

AIR JET PUMP

The air jet pump located on the rear pressure bulkhead extracts air from the outflow/safety valves during normal operation. This pump, operated by engine bleed air, ensures that the desired differential between the outflow valve reference pressure and cabin pressure is maintained.

PRESSURE REGULATORS

The two pressure regulators function as safety devices if there is a loss of cabin air pressure. The regulators, located on the rear pressure bulkhead, shut both outflow/safety valves when the cabin altitude reaches 13,500 ±1,500 feet, thus minimizing a further drop in cabin pressure.

FAN-OPERATED VENTURI

The fan-operated venturi is also located on the rear pressure bulkhead. It is controlled by the PRESSN switch (Figure 12-2) which is located on the roof panel.

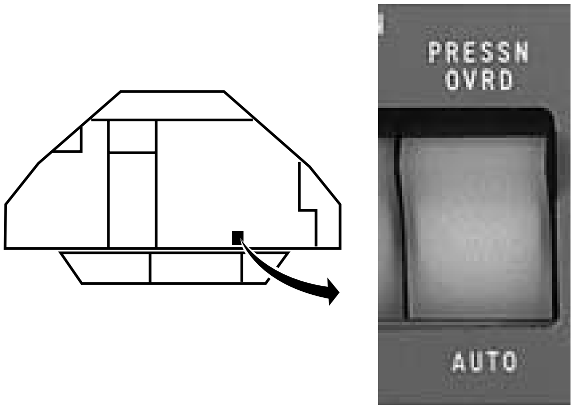


Figure 12-2. PRESSN Switch

With the PRESSN switch in the AUTO position and the airplane on the ground, the venturi operates (fan running) if an engine or APU air valve is open or for approximately 20 seconds after the aircraft has landed. With the weight off the wheels (in flight), the fan stops, thus allowing the cabin to be pressurized. With the switch in the OVRD position and the airplane on the ground, the fan stops, the venturi ceases operation, thus allowing the cabin to be pressurized.

CONTROLS AND INDICATORS

In addition to the CABIN and the RATE knobs on the CABIN CONTROLLER and the PRESSN switch on the roof panel, the pressurization system incorporates the following controls and indicators:

- CABIN ALTITUDE warning annunciator
- CABIN triple indicator
- Warning horn
- HORN ISOLATE button
- MANUAL CABIN ALTITUDE CONTROL knob
- PRESSURIZATION CONTROL lever
- DUMP VALVE lever

CABIN ALTITUDE Warning Annunciator and Horn

The CABIN ALTITUDE warning annunciator (Appendix B) and warning horn alert the crew that the cabin has exceeded an altitude of 9,300 feet.

When this occurs, the following takes place:

- At 9,300 ±300 feet (cabin altitude), the CABIN ALTITUDE warning annunciator lights and the warning horn sounds. The horn may be silenced by depressing the HORN ISOLATE pushbutton located on the No. 1 throttle lever (Figure 12-3).

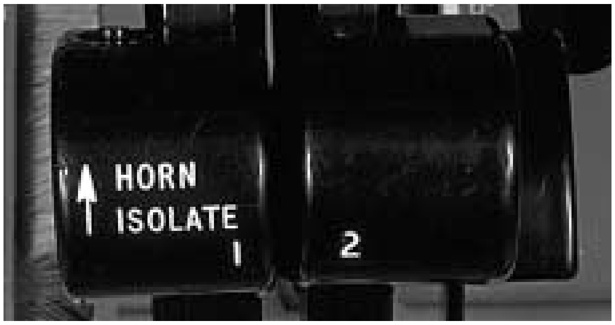


Figure 12-3. HORN ISOLATE Pushbutton

The warning annunciator is located on the master warning panel.

NOTE

With flaps selected beyond 15°, the warning horn is electrically isolated for this condition.

- At 13,500 ±1,500 feet (cabin altitude), the pressure regulators cause the outflow/safety valves to close, minimizing further loss of cabin pressure through the valves.

In order to stabilize any of the above conditions, the pilot can select manual pressurization control or use the auxiliary heating valve to provide emergency pressurization. The dump valve can be used to relieve excess pressure in the airplane.

CABIN Triple Indicator

A triple pointer CABIN indicator (Figure 12-4) is installed on the copilot's instrument panel. The A pointer displays cabin altitude, the plain (unmarked) pointer indicates cabin altitude rate of change, and the red and white barber-pole pointer shows cabin differential pressure.

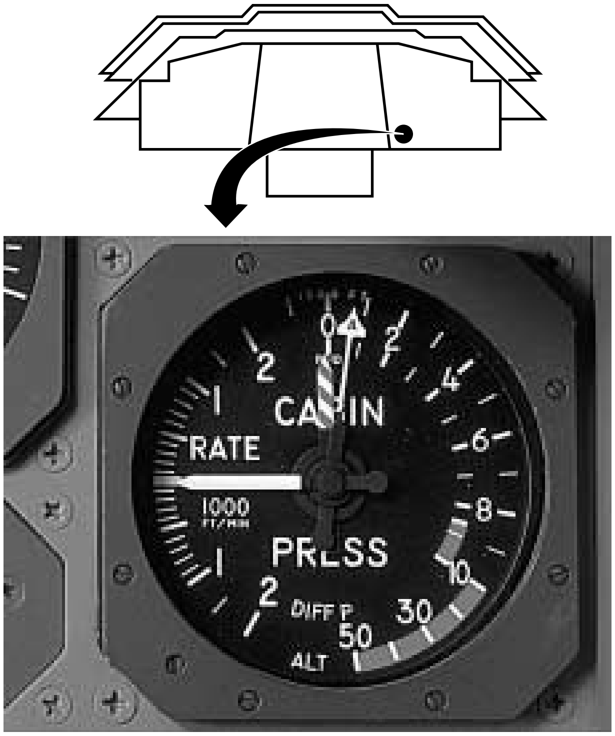


Figure 12-4. CABIN Triple Indicator

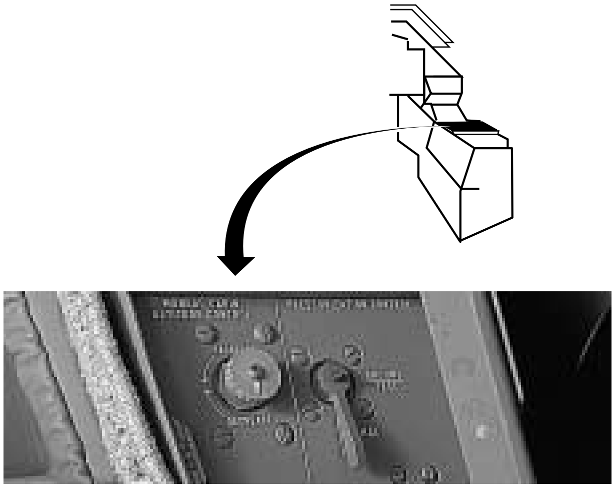


Figure 12-5. Manual Pressurization Controls

MANUAL CABIN ALTITUDE CONTROL Knob and PRESSURIZATION CONTROL Lever

The MANUAL CABIN ALTITUDE CONTROL knob and the PRESSURIZATION CONTROL lever (Figure 12-5) allow the copilot to manually control the pressurization system. The manual controls are located on the copilot's side console.

Control of one outflow/safety valve and the desired cabin altitude and rate of change is accomplished with the MANUAL CABIN ALTITUDE CONTROL knob.

To ensure a smoother changeover from normal (automatic) operation, the copilot should slowly turn the MANUAL CABIN ALTITUDE CONTROL knob until cabin altitude increases slightly, then slowly select the GROUND TEST position on the PRESSURIZATION CONTROL lever. If manual control is ineffective, use the dump valve to control pressurization.

DUMP VALVE Lever

The DUMP VALVE lever (Figure 12-6), located on the right side of the center control pedestal, can be used to open or close a butterfly valve in a duct between the vestibule and nose gear bay. The lever can be positioned anywhere in between the OPEN and SHUT positions to vary the exhausting of cabin air. With the DUMP VALVE lever in the full OPEN position, it operates in conjunction with the ram-air valve to provide ventilation when the cabin is unpressurized during flight.

OPERATION

BEFORE STARTING ENGINES

Check that the DUMP VALVE lever is shut, and that the PRESSURIZATION CONTROL lever is in the NORMAL flight position, and that the MANUAL CABIN ALTITUDE CONTROL is in the full DECREASE position. Set the F/DK VLV switch to CLOSE, CABIN FLOOD switch to CLOSE, and the PRESSN switch to AUTO. Check that both MAIN AIR VLV switches are at CLOSE and both MAIN AIR VALVE annunciators are extinguished. Preset the CABIN TEMP selector to the required temperature on the AUTO scale.

If cruising at altitudes above 25,000 feet, rotate the pressure controller CABIN knob to set the inner scale of the CABIN CONTROLLER to cruise altitude +2,000 feet.

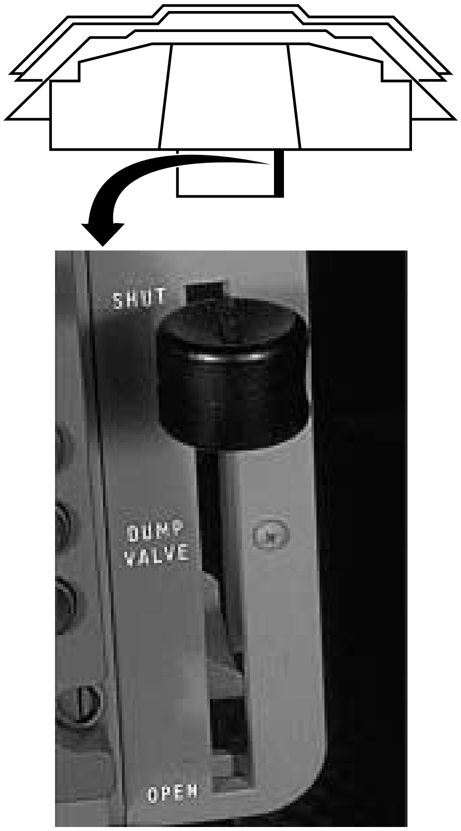


Figure 12-6. DUMP VALVE Lever

If cruising at altitudes below 25,000 feet, set the cabin pointer to 2,000 feet above departure airfield altitude.

In all cases the selected cabin altitude must be at least 1,000 feet above airfield altitude.

Set the RATE knob at midtravel.

AFTER STARTING ENGINES

Introduce air circulation by setting both MAIN AIR VLV switches to OPEN; check that both MAIN AIR VALVE annunciators are on, or APU AIR VLV to OPEN. Adjust the temperature selector if required. The F/DK VLV switch can be used for rapid heating of the flight deck.

TAKEOFF

Before beginning the takeoff, the MAIN AIR VLV switches must be set to CLOSE. Check that both MAIN AIR VALVE annunciators are extinguished. Check that the F/DK VLV switch is set to CLOSE. Check that the PRESSN switch is set to AUTO.

AFTER TAKEOFF

As soon as practicable after takeoff, position the MAIN AIR VLV switches to OPEN. Check that MAIN AIR VALVE annunciators are extinguished. Check that the DUCT OVHT ↑ annunciator remains extinguished. Adjust the cabin RATE of change knob to obtain the desired rate of climb appropriate to the top of climb height.

If the airplane fails to pressurize during the climb, it may be due to an auto mode malfunction. Select the PRESSN switch to OVRD; this will ensure that the fan-oprated venturi, normally operating on the ground to hold the outflow valves open, is switched off.

In the event of cabin altitude not being controlled satisfactorily by the cabin pressure control, turn the MANUAL CABIN ALTITUDE CONTROL slightly in the INCREASE direction. The PRESSURIZATION CONTROL lever should then be turned to the GROUND TEST position after which the MANUAL CABIN ALTITUDE CONTROL can be used to attain the required cabin altitude.

DURING FLIGHT

Check that the cabin altitude and temperature are maintained satisfactorily. Crew compartment temperature may be increased independently of the cabin by operation of the F/DK VLV switch.

DESCENT

At the start of descent, set the cabin pressure control according to Table 12-1 for the destination airfield. Adjust the cabin RATE of change knob to give the desired rate of descent. This should not exceed 300 ft/minute.

BEFORE LANDING

The cabin altitude setting for landing depends upon the destination barometric pressure (QNH). Apply the correction given in Table 12-1 for the appropriate barometric pressure (QNH) to the airfield elevation. Then set that value on the outer scale of the CABIN CONTROLLER.

Close the main air valves before the aircraft descends below 250 feet to reduce engine

Table 12-1. CABIN ALTITUDE SETTING FOR LANDING

QNH—MB	970	975	980	985	990	995	1,000
IN. Hg	28.64	28.79	28.94	29.09	29.23	29.38	29.53
CORRECTION —FEET	+ 1,500	+ 1,350	+ 1,250	+ 1,100	+ 950	+ 800	+ 650
QNH—MB	1,005	1,010	1,015	1,020	1,025	1,030	1,035
IN. Hg	29.68	29.82	29.97	30.12	30.27	30.41	30.56
CORRECTION —FEET	+ 550	+ 400	+ 250	+ 100	0	– 150	– 300

bleed air and thereby providing available maximum thrust in the event of a missed approach/go around manuever.

Ensure cabin differential pressure is at zero before touchdown.

ELECTRICAL POWER SOURCES

DC POWER

PE bus bar

- Cabin altitude warning
- Ram-air valve control

PS2 bus bar

- Recirculating fan control
- Flood-flow valve actuator

AC POWER

XS2 bus bar

- Fan-operated venturi

LIMITATIONS

The following limitations pertain to the Hawker 800 XP airplanes' pressurization system:

- Maximum cabin differential pressure is 8.55 psi.
- Maximum allowable negative pressure differential is –0.5 psi.
- Safety valves prevent pressure buildup in excess of 8.6 to 8.8 psi.

QUESTIONS

1. The pressurization system is capable of maintaining a cabin altitude of 8,000 feet at a maximum cabin differential pressure when the airplane's actual altitude is:

A. 31,000 feet
B. 41,000 feet
C. 43,000 feet
D. 35,000 feet

2. The pressurization system can maintain a maximum cabin differential pressure of:

A. 7.35 psi
B. 8.55 psi
C. 9.35 psi
D. 5.38 psi

3. The maximum allowable negative cabin differential pressure is:

A. 0.5 psi
B. 8.55 psi
C. 1.25 psi
D. 0.025 psi

4. With the airplane on the ground and the air-conditioning system on, the PRESSN switch prevents cabin pressurization when in the:

A. OVRD position
B. ON position
C. OFF position
D. AUTO position

5. If the pressurization system fails, excessive cabin pressure is prevented by the:

A. Outflow/safety valve
B. Dump valve
C. Pressure controller
D. Ram-air valve

CHAPTER 13

HYDRAULIC POWER SYSTEMS

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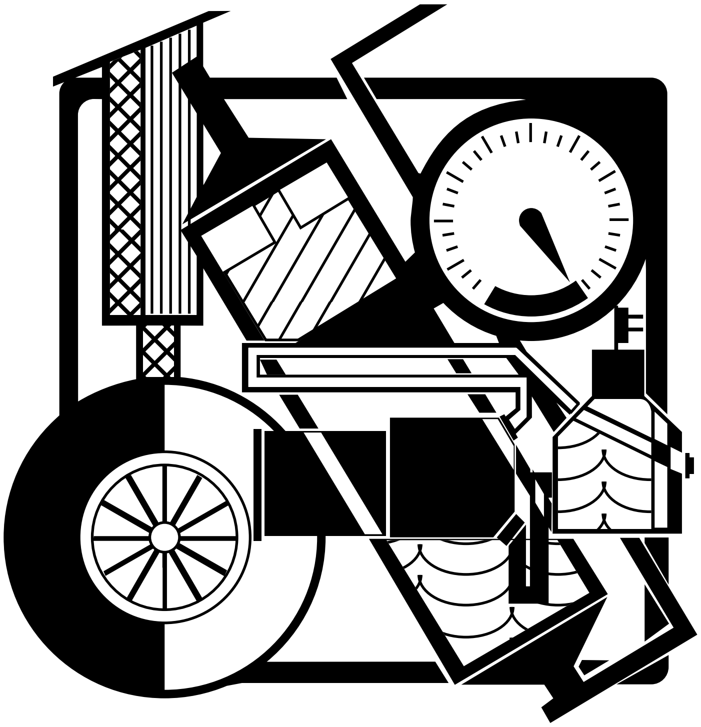
CHAPTER 13
HYDRAULIC POWER SYSTEMS



INTRODUCTION

The Hawker-800 XP airplane has both main and auxiliary hydraulic systems. The main system is pressurized by two engine-driven pumps, one on each engine. The system provides pressure for actuation of the landing gear, flaps, airbrakes, nosewheel steering, brakes, stick pusher, and thrust reversers. System operation is monitored by annunciators and a system pressure gage.

The auxiliary hydraulic system, pressurized by a hand pump, provides pressure only for emergency extension of the landing gear and for flap operation.

GENERAL

The output of the variable-volume, engine-driven pumps is manifolded to pressurize the main hydraulic system. Either pump is capable of actuating all the subsystems. A hand pump is provided in the rear equipment bay for ground pressurization of the system.

A second hand pump (in the cockpit) receives fluid from a separate tank for emergency extension of the landing gear and for flap operation. Auxiliary system tank fluid level is monitored by an amber annunciator.

MAIN HYDRAULIC SYSTEM

GENERAL

Controls and indicators related to hydraulic power are shown in Figure 13-1.

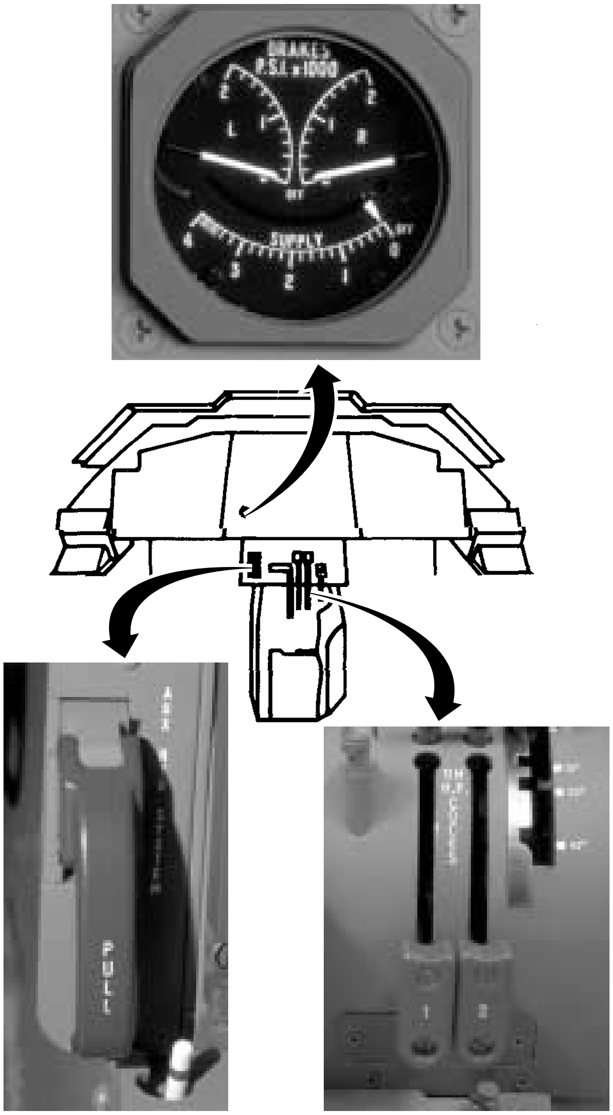


Figure 13-1. Hydraulic Systems Controls and Indicators

The main system reservoir (2.4 U.S. gallon capacity) is pressurized by regulated engine bleed air to prevent foaming, cavitation, and to assure an adequate supply of MIL-H-5606C fluid to the pumps under all operating conditions (Figure 13-2). Flow to the pumps passes through a suction filter that bypasses when clogged, and through on-off valves interconnected with the HP COCK levers. The levers are selected open on engine start. They are selected closed on engine shutdown. With the on-off valve closed, a closed circuit provides a supply at the pump for lubrication purposes.

Low-pressure output of either pump is indicated by the illumination of the applicable annunciator (HYD 1 LO PRESS or HYD 2 LO PRESS) (Table 13-1). Fluid temperatures in excess of 90° C are indicated by illumination of the HYD OVHT annunciator. All colored annunciators are shown in Appendix B of this manual.

TABLE 13-1. Indicators and Annunciators

Item	Location	Parameter
Combined BRAKES (2) and SUPPLY indicator.	Center instrument panel.	psi Brakes: 2,000 Supply: 3,000
HYD OVHT	Main MWS panel.	Hydraulic fluid overheat warning 90° C.
HYD 1 LO PRESS HYD 2 LO PRESS	Main MWS panel.	Main supply low pressure warning ≤ 1,500 psi.
EMER BRK LO PRESS	Main MWS panel.	Emergency brakes accumulator low pressure warning ≤ 2,250 psi.
Accumulator pressure indicators (main and wheel brakes).	Rear equipment bay.	Initial charge at 1,000 psi.
Accumulator pressure indicators (thrust reversers).	Rear equipment bay.	Initial charge at 1,500 psi.
Main reservoir contents level tube.	Rear equipment bay.	Parameters stated on label next to reservoir.
Main reservoir air pressure gage.	Rear equipment bay.	0 to 18 psi is normal.

Pressurized fluid flows through a pressure filter prior to entering the subsystems. The pressure filter incorporates a pop-out button that extends if the element has an increasing differential pressure caused by clogging and does not have a relief valve.

The electrically operated DC hydraulic pressure indicator registers main system pressure on the SUPPLY gage. Loss of DC power causes the indicator pointer to move to the OFF position.

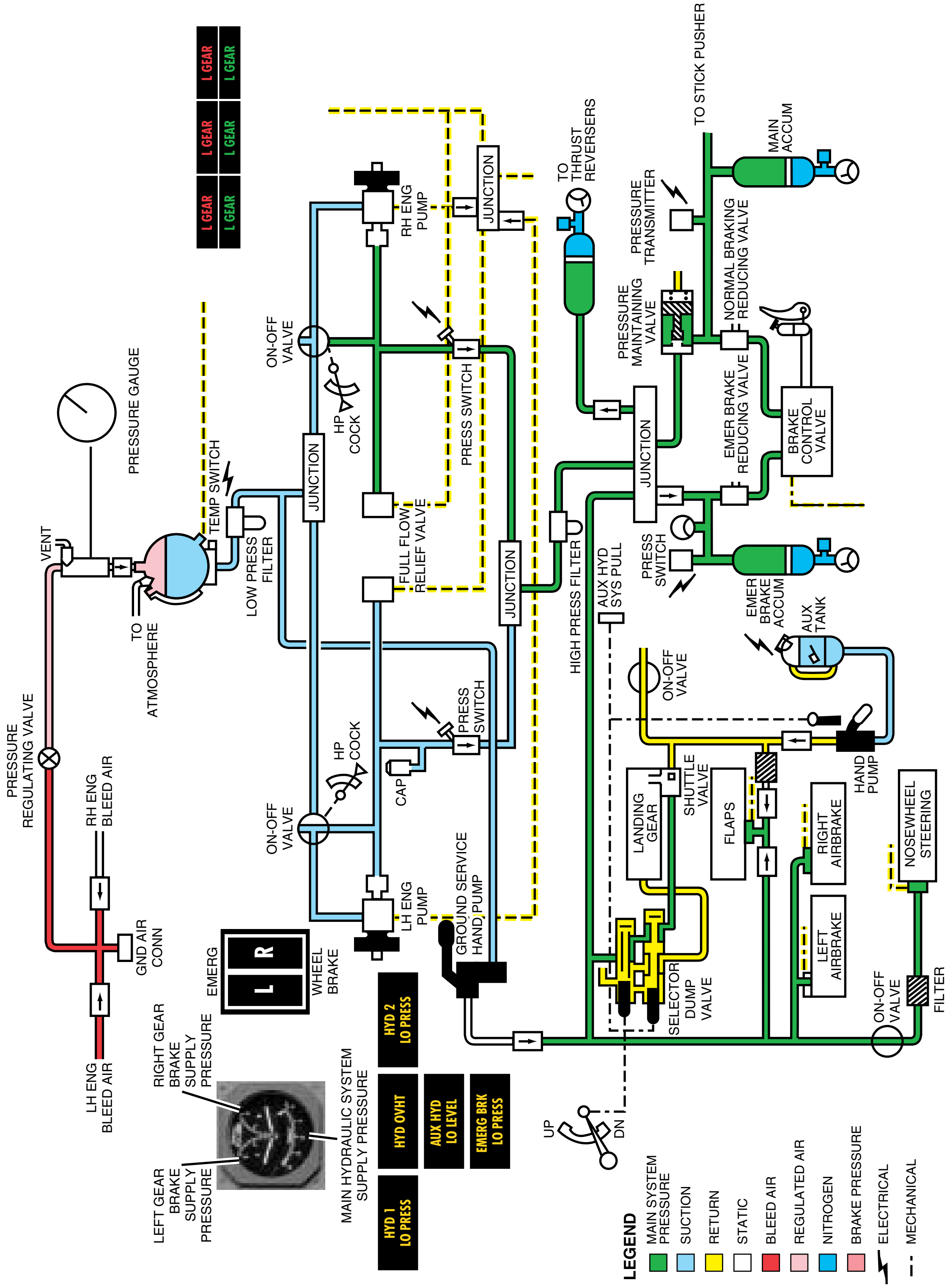


Figure 13-2. Hydraulic System

With the airbrakes shut, proper reservoir fluid level can be determined by sighting the fluid level sight gage against the marks on the adjacent indicating plate (Figure 13-3).

Air supply low pressure is indicated by the HYD TANK pressure gage (Figure 13-5). The gage should indicate approximately 15 psi.

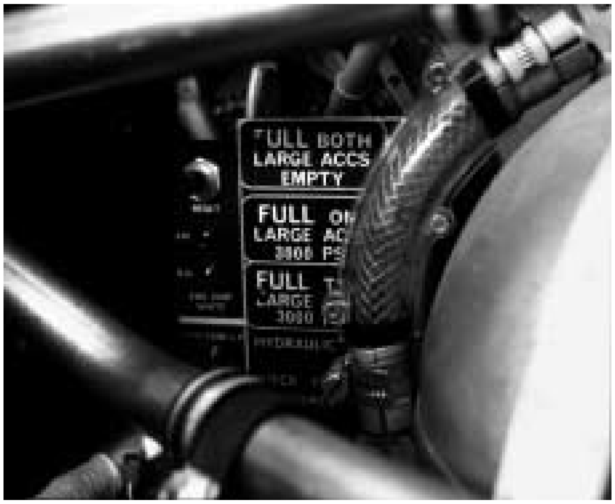


Figure 13-3. Reservoir Pressurization and Fluid Level Indication

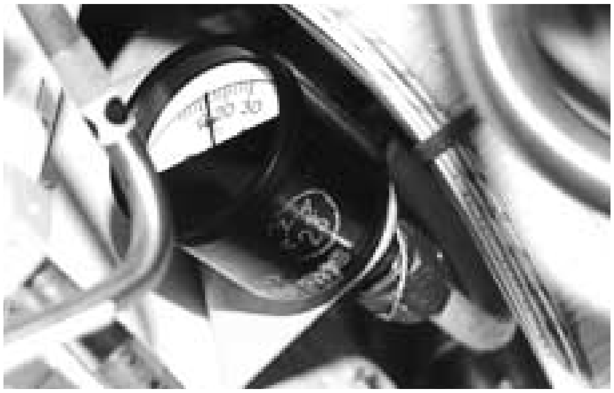


Figure 13-5. Reservoir Pressure Gage

OPERATION

When the right engine is started, the pump receives fluid through the suction filter and the open on-off valve. As pressure increases to 1,750 psi, the amber HYD 2 LO PRESS annunciator goes out (Figure 13-2). Increasing pressure from the pump opens the pressure-maintaining valve, and pressure is indicated on the supply scale of the pressure indicator. Normal system pressure will stabilize at 3,000 psi. The system is now capable of actuating all subsystems.

As the left engine is started, the pressure output of the left pump is added to the system (Figure 13-2). The amber HYD 1 LO PRESS annunciator goes out as pressure increases. The system is now operating at full potential.

If either pump fails, the low pressure switch operates when the pressure has fallen to 1,500 psi. The contact of the low pressure switch causes the HYD 1(2) LO PRESS annunciator on the roof panel to be lit, after an initial delay of 57 seconds to allow the pressure to recover should the low pressure condition be due to system fluctuation rather than a malfunction.

If both pumps fail, or if system fluid is lost, both low-pressure annunciators will come on, but the pressure indicator will show 2,300 psi (the pressure-maintaining valve has closed, trapping pressure in the main accumulator for normal brake operation). The landing gear and flaps can be lowered with the auxiliary hydraulic system.

Two full-flow relief valves are provided, one in each pressure line. In the event of pump failure (loss of pump pressure control), the associated relief valve opens at approximately 4,000 psi to return fluid to the reservoir.

Leakage from any pressure line, suction line, or system component will deplete system fluid. Since there is no indication of decreasing fluid level, fluid loss may not be noticed until the pumps cavitate and the annunciators come on. Leakage within a subsystem may be detected by an abnormal drop in pressure as that subsystem is actuated.

In the event system fluid overheat is indicated by illumination of the HYD OVHT annunciator, continue the flight. The fault must be corrected prior to the next flight.

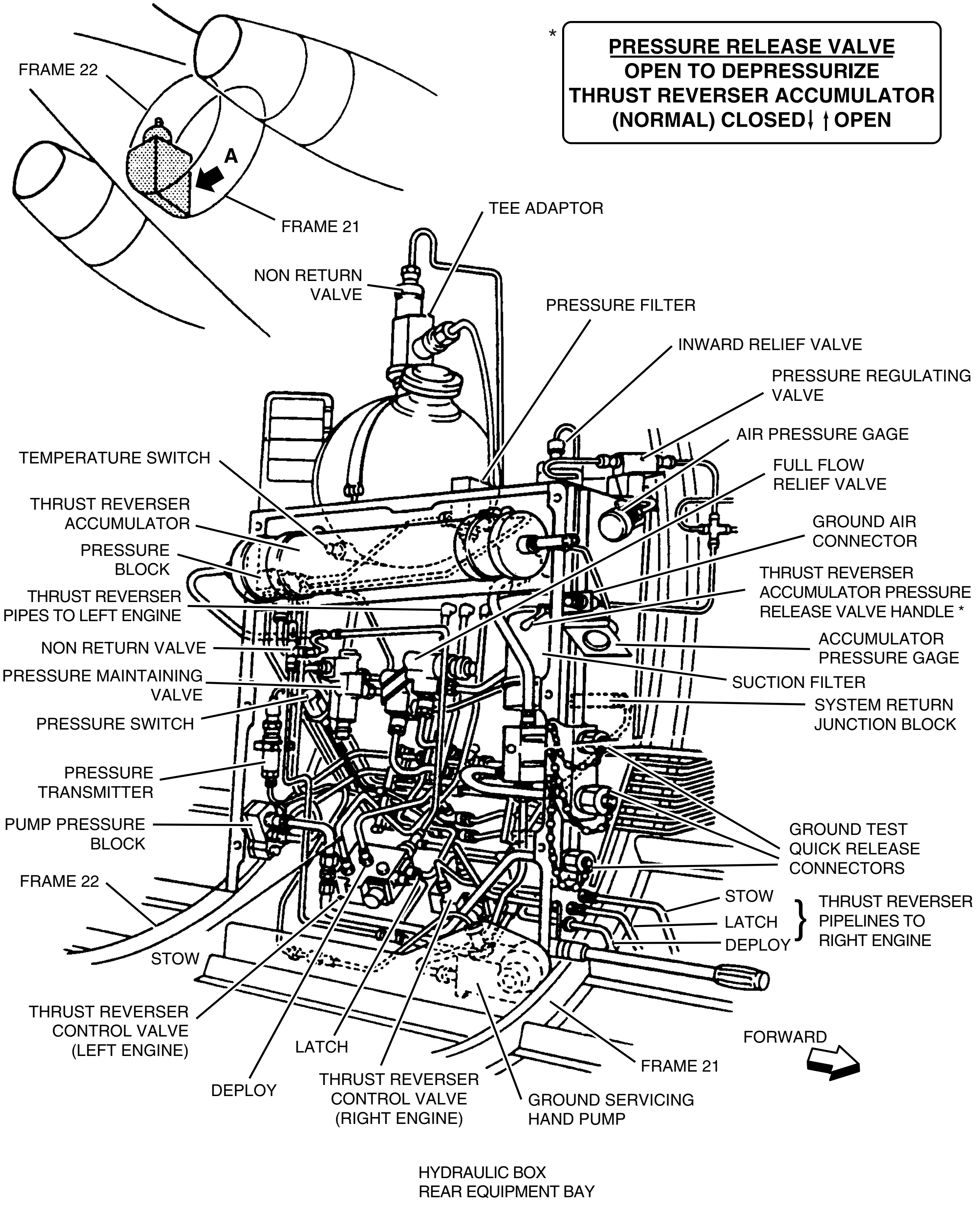

*
PRESSURE RELEASE VALVE
OPEN TO DEPRESSURIZE
THRUST REVERSER ACCUMULATOR
(NORMAL) CLOSED↓ ↑OPEN
FRAME 22
A
FRAME 21
TEE ADAPTOR
NON RETURN VALVE
PRESSURE FILTER
INWARD RELIEF VALVE
PRESSURE REGULATING VALVE
AIR PRESSURE GAGE
TEMPERATURE SWITCH
FULL FLOW RELIEF VALVE
THRUST REVERSER ACCUMULATOR
GROUND AIR CONNECTOR
PRESSURE BLOCK
THRUST REVERSER ACCUMULATOR PRESSURE RELEASE VALVE HANDLE *
THRUST REVERSER PIPES TO LEFT ENGINE
ACCUMULATOR PRESSURE GAGE
NON RETURN VALVE
PRESSURE MAINTAINING VALVE
SUCTION FILTER
SYSTEM RETURN JUNCTION BLOCK
PRESSURE SWITCH
PRESSURE TRANSMITTER
PUMP PRESSURE BLOCK
GROUND TEST QUICK RELEASE CONNECTORS
FRAME 22
STOW
LATCH
DEPLOY
THRUST REVERSER PIPELINES TO RIGHT ENGINE
STOW
THRUST REVERSER CONTROL VALVE (LEFT ENGINE)
FORWARD
LATCH
FRAME 21
DEPLOY
THRUST REVERSER CONTROL VALVE (RIGHT ENGINE)
GROUND SERVICING HAND PUMP
HYDRAULIC BOX
REAR EQUIPMENT BAY

Figure 13-4. Main Hydraulic System Components

AUXILIARY HYDRAULIC SYSTEM

GENERAL

The auxiliary hydraulic system is pressurized by the hand pump located on the floor, at the right side of the pilot. The removable pump handle, stowed on the front of the copilot's seat base, can also be used as the rudder gust lock. MIL-H-5606C fluid is supplied by a separate 6 U.S. pint tank in the nose gear bay (Figure 13-6).

Fluid level is observed in the transparent sight gage. A float switch in the tank is connected to the amber AUX HYD LO LEVEL annunciator. If fluid level is reduced by one-half pint or more, the annunciator will illuminate.

The auxiliary hydraulic system provides an alternative source of hydraulic power for landing gear lowering and flaps operation.

The system is hand-operated, and when selected, is independent of the main system.

Emergency lowering of the landing gear is available regardless of the setting of the landing gear selector lever but the flaps must be selected to the required setting.

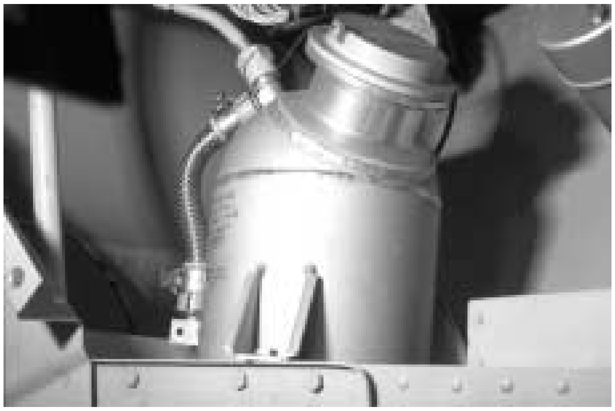


Figure 13-6. Auxiliary Hydraulic Tank

When using the auxiliary system, lowering the landing gear always occurs before operation of the flaps. The system cannot be used to raise the landing gear.

A full reservoir provides sufficient fluid for at least one emergency lowering of the landing gear followed by at least one extension of the flaps to the landing position.

CAUTION

If the auxiliary system selection has been made following a main system failure, the auxiliary system must not be reset in flight.

Ignoring this CAUTION could result with the landing gear collapsing.

TABLE 13-2. Controls and Indications

Control	Location
AUX HYD SYSTEM PULL selector handle	Left side of center instrument panel, adjacent to FMS No. 1 display.
EMERG HYD PUMP operating handle socket The rudder gust lock serves as the handle.	Left side of center instrument panel, adjacent to FMS No. 1 display, beneath the AUX HYD SYSTEM PULL selector handle.
AUX HYD LO LEVEL	MWS main panel
Auxiliary reservoir contents level indicator tube	Visible from nose gear bay.

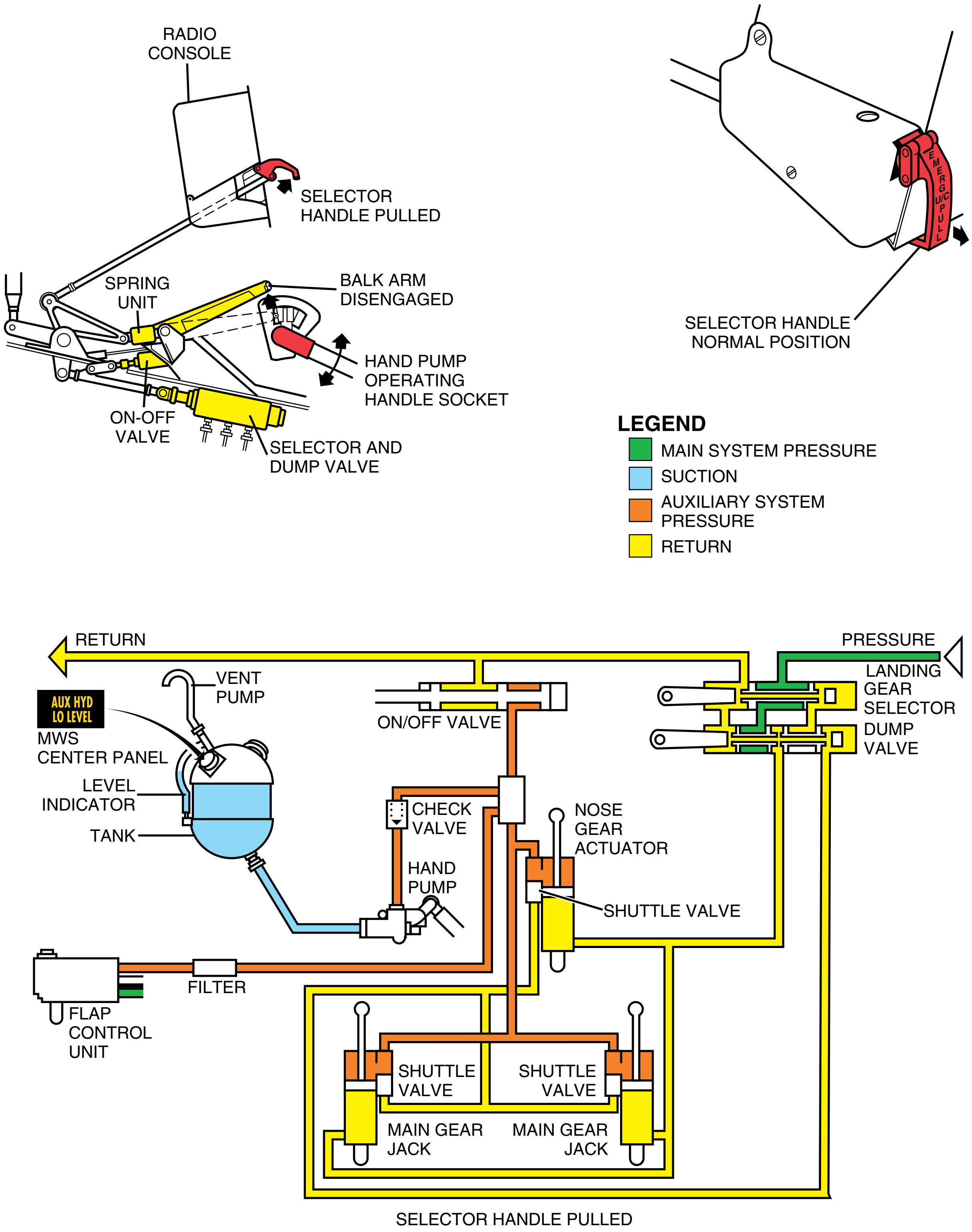


Figure 13-7. Auxiliary Hydraulic System Schematic

OPERATION

Operation of the auxiliary hand pump in normally blocked by a balk arm. To prepare the system for operation, pull the AUX HYD SYSTEM lever (Figure 13-7). This action:

- Disengages the balk arm, freeing the hand pump for operation
- Closes the on-off valve, allowing hand pump pressure buildup
- Positions the dump valve to block main system pressure and to connect landing gear actuators to the reservoir return

Operation of the hand pump will now create pressure for emergency extension of the landing gear and for flap operation, when selected.

CAUTION

When operating the auxiliary hydraulic system to lower the landing gear, the hand pump action must be continued, after three greens are achieved, until positive resistance is felt to make sure the landing gear is down and locked.

HYDRAULIC SUBSYSTEMS

Landing gear, flaps, airbrakes, nosewheel steering, brakes, stick pusher, and thrust reversers are hydraulically powered. The application of hydraulic power is presented in Chapter 14, "Landing Gear and Brakes," in Chapter 15, "Flight Controls," and in Chapter 7, "Powerplant," for thrust reversers.

ELECTRICAL POWER SOURCES

PE bus bar

- HYD 1 LO PRESS annunciator
- HYD 2 LO PRESS annunciator
- AUX HYD LOW LEVEL annunciator
- EMER BRAKE LOW PRESSURE annunciator

PS2 bus bar

- Main SUPPLY pressure indicator
- HYD OVHT annunciator

QUESTIONS

1. If one engine-driven pump fails:
   A. The hydraulic pressure indicator will show 3,000 psi, and the applicable low-pressure annunciator will come on.
   B. The hydraulic pressure indicator will show 2,300 psi, and the applicable low-pressure annunciator will come on.
   C. The hydraulic pressure indicator will show 3,000 psi, and neither low-pressure annunciator will be on.
   D. The hydraulic pressure indicator will show 4,000 psi.

2. If both engine-driven pumps fail:
   A. Indicated pressure is 3,000 psi, and both low-pressure annunciators are on.
   B. Indicated pressure is 2,300 psi, and both low-pressure annunciators are on.
   C. No pressure is indicated, and both low-pressure annunciators are on.
   D. None of the above

3. The main hydraulic reservoir is pressurized to prevent:
   A. Pump cavitation
   B. Hydraulic fluid overheat
   C. Loss of wheel brakes
   D. All of the above

4. The first step in operating the auxiliary hydraulic system is:
   A. Dump main system pressure.
   B. Operate the auxiliary hand pump.
   C. Pull the AUX HYD SYSTEM lever.
   D. Check that the AUX HYD LO LEVEL annunciator is not on.

5. The purpose of the pressure-maintaining valve in the main hydraulic system is:
   A. To maintain 2,300 psi in the main accumulator for operation of the emergency brakes
   B. To maintain 2,300 psi in the emergency brake accumulator for emergency brake operation
   C. To maintain 3,000 psi in the main accumulator for normal brake operation
   D. To maintain 2,300 psi in the main accumulator for normal brake operation

6. If one engine-driven pump fails:
   A. The remaining pump is capable of actuating all subsystems.
   B. System pressure will be insufficient to actuate all subsystems.
   C. The auxiliary system will provide pressure for actuation of all subsystems.
   D. Pressure in the system will be cut in half.

CHAPTER 14
LANDING GEAR AND BRAKES

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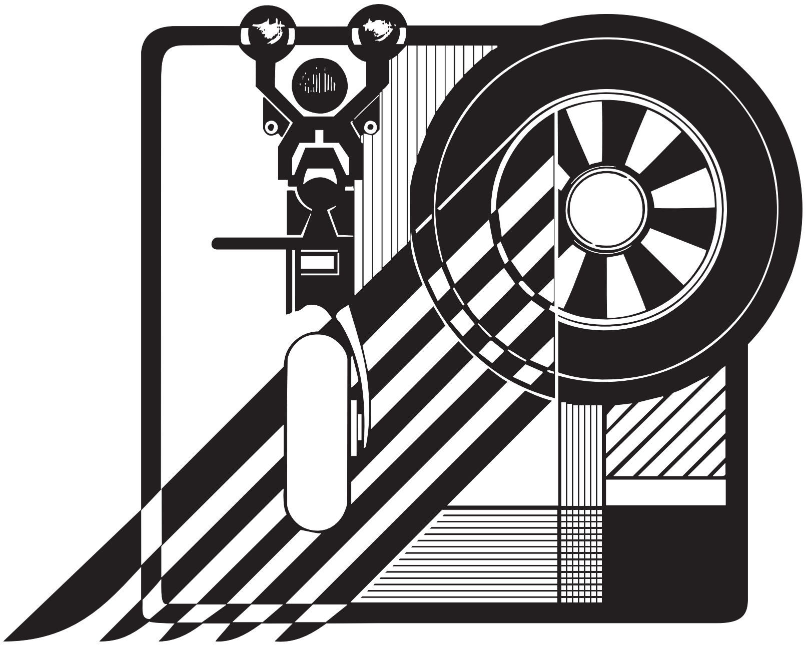
CHAPTER 14
LANDING GEAR AND BRAKES



INTRODUCTION

The hydraulically actuated landing gear on the Hawker-800 XP airplane is enclosed by hydraulically and mechanically actuated doors. Gear position and warning are provided by indicator lights and a warning horn.

Nosewheel steering is also hydraulically powered. It provides directional control on the ground. The brake system, which is controlled by master cylinder pressure, provides for normal and emergency braking. Parking brakes are applied with a lever in the cockpit.

GENERAL

The inward retracting main gear and the forward retracting nose gear are operated by one hydraulic actuator attached to each gear. The nose gear doors are mechanically operated by movement of the gear. The main gear inboard doors are hydraulically operated; the outboard doors are mechanically operated.

Gear position indication is provided by six indicator lights on the center instrument panel and a nosegear mechanical indicator pin on the center pedestal. Standby indicator lights for the main gear are located on the right side console. In addition, a warning horn sounds if all three gear are not down and locked when flap position and/or throttle settings are in the landing configuration.

The hydraulically actuated nosewheel steering system, manually controlled by a handwheel on the left console, provides steering 45° right and left of center. The steering actuator also functions as a shimmy damper. A mechanically operated shutoff valve provides hydraulic pressure to the system only when the nose gear is down and locked.

Master cylinders operated by the tops of the rudder pedals provide controlling pressure to actuate brake control valves for normal or emergency operation of the brakes. Selection of normal, emergency, or parking brakes is made with the wheel brake lever in the cockpit. Antiskid braking is available only during normal braking.

LANDING GEAR

MAIN GEAR

The main gear (Figures 14-1 and 14-2) includes a shock strut (oleo-pneumatic leg), torque link, hydraulic gear actuator (jack), side stay, wheels and brake assemblies, inboard door, and outboard door (fairing). Each wheel contains fusible plugs that release air from the tire in the event of excessive pressure buildup due to heat. The side stay is the rigid component of the assembly when the gear is extended or retracted. The gear is attached to the wing structure at the trunnion points. The outboard door is mechanically actuated by gear movement; the inboard door is hydraulically actuated and is closed with the gear extended or retracted. The maximum permissible airspeed for extending or retracting the landing gear and with the gear extended is 220 knots IAS.

NOSE GEAR

The nose gear (Figures 14-3 and 14-4) includes a shock strut, torque link, hydraulic gear actuator (jack), drag brace, and the necessary mechanical linkage for opening and closing the three gear doors.

The drag brace is the stabilizing component when the gear is extended. Gear actuation and door movement are accomplished with one hydraulic actuator. The gear is self-centering when the shock strut is fully extended. The steering disconnect pin must be installed prior to taxiing.

All three nose gear doors are mechanically actuated by gear movement. The forward doors are closed with the gear extended or retracted; the aft door remains open with the gear extended. The forward doors can be manually

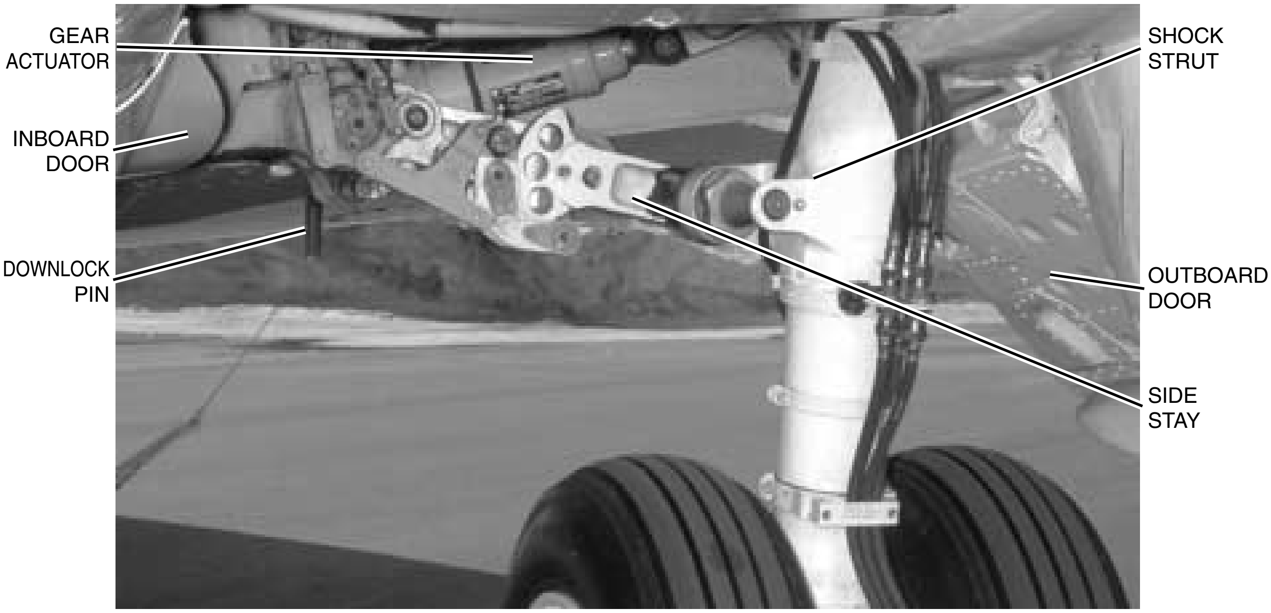


Figure 14-1. Main Landing Gear

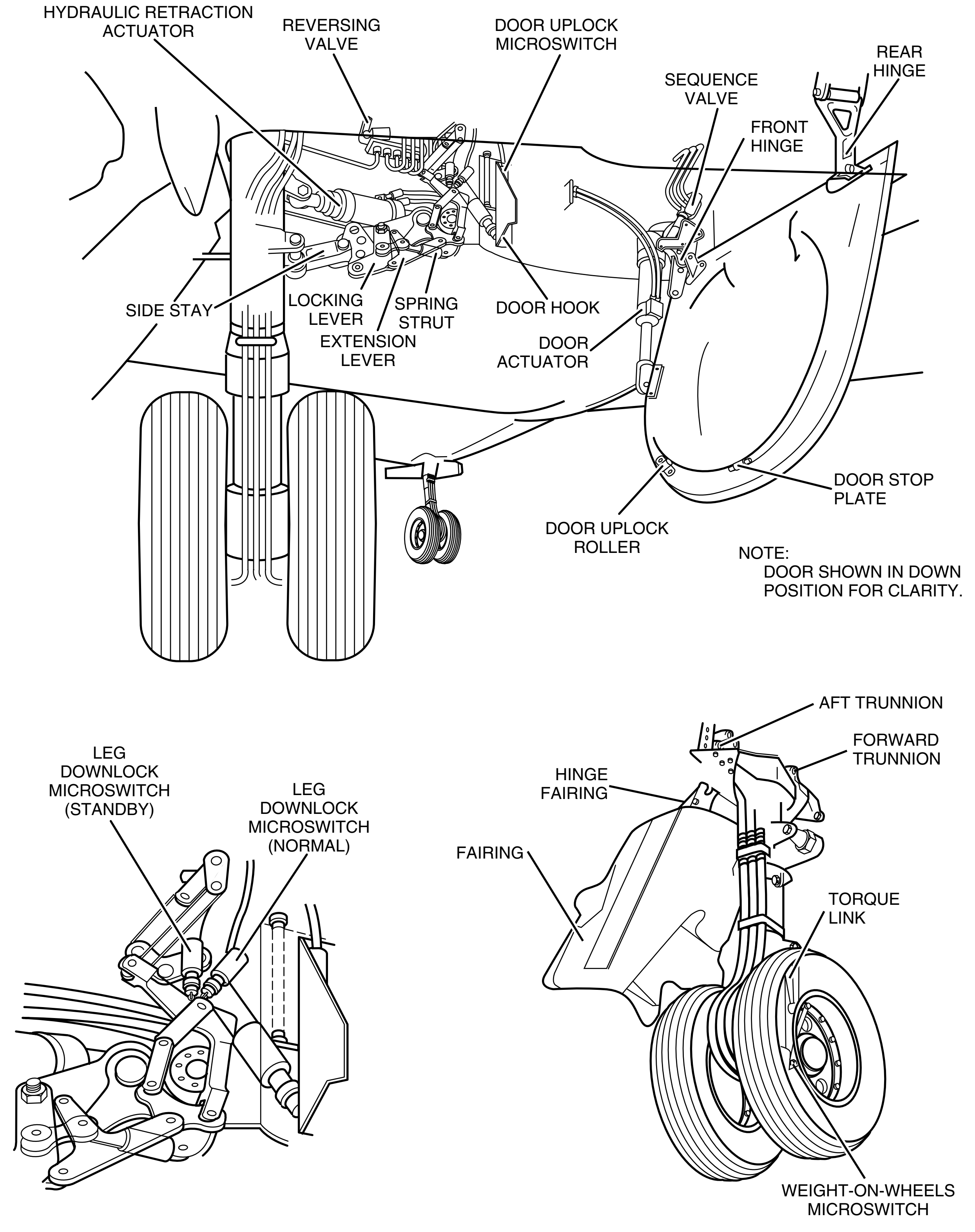
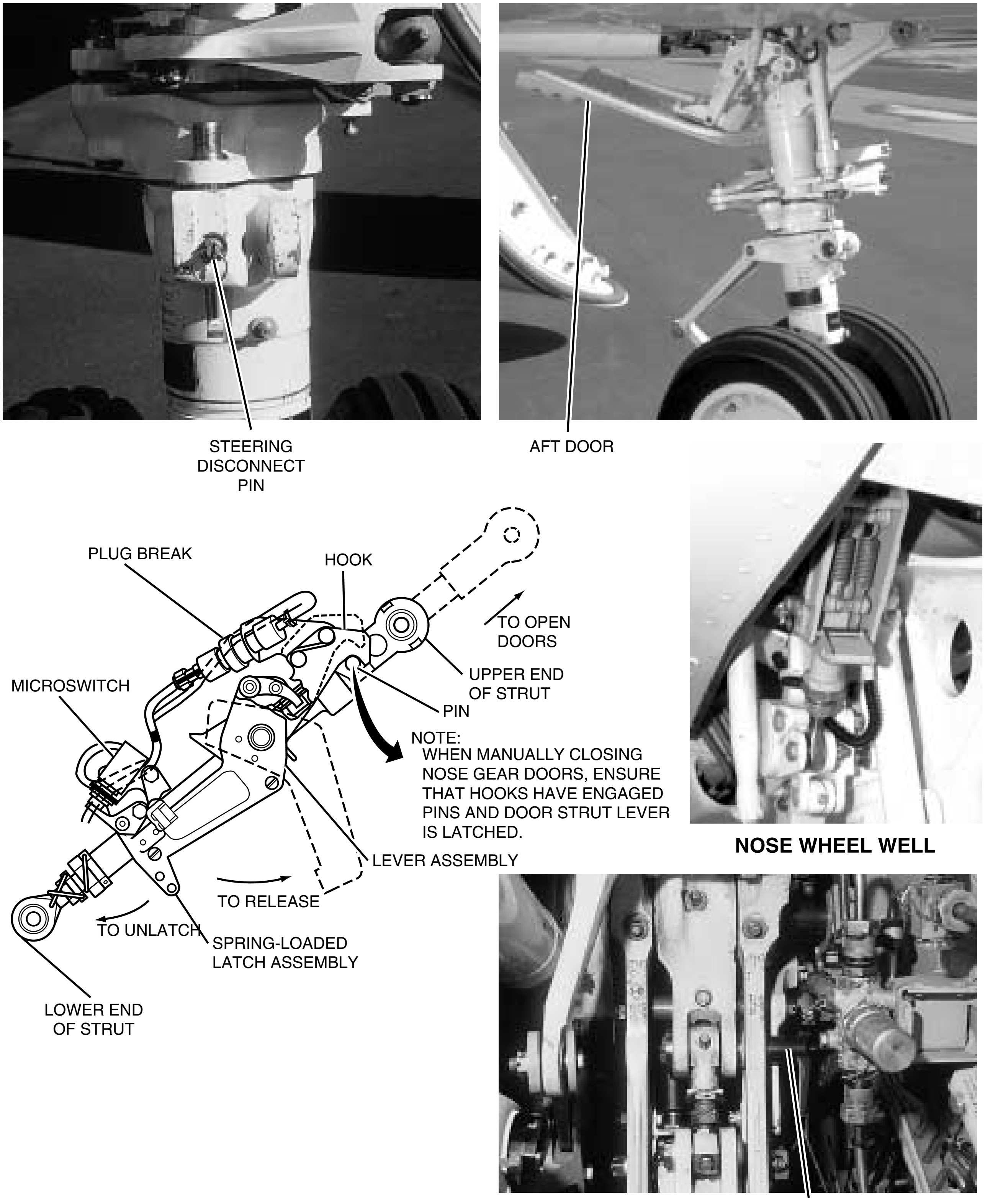


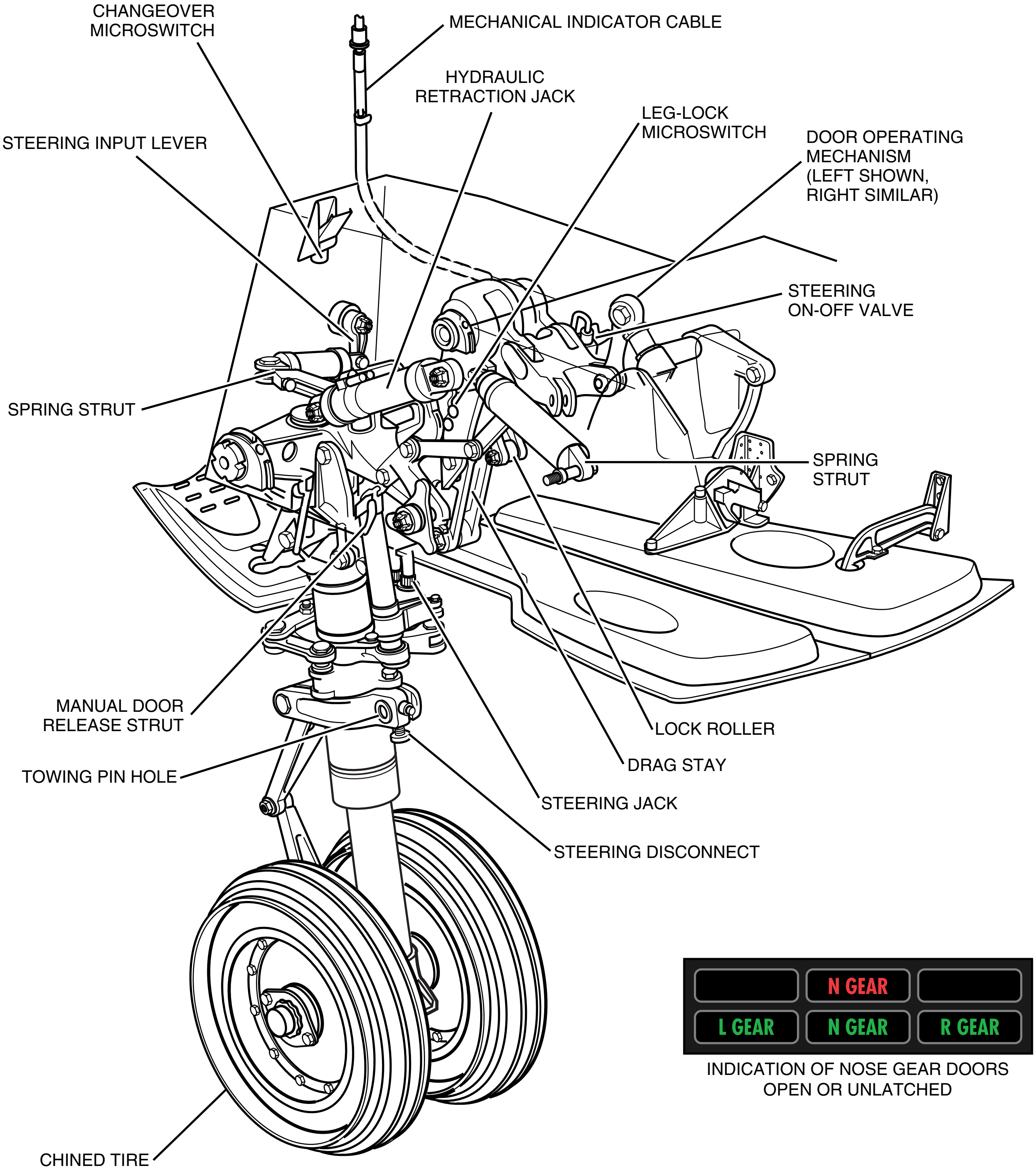
Figure 14-2. Main Gear Components



NOTE:
WHEN MANUALLY CLOSING NOSE GEAR DOORS, ENSURE THAT HOOKS HAVE ENGAGED PINS AND DOOR STRUT LEVER IS LATCHED.

NOSE WHEEL WELL

Figure 14-3. Nose Landing Gear



INDICATION OF NOSE GEAR DOORS OPEN OR UNLATCHED

Figure 14-4. Nose Gear Components

opened on the ground with a door release lever (Figure 14-3). The doors must be manually closed and the release lever latched prior to flight.

A microswitch on the door release lever illuminates the red N GEAR position annunciator (Figure 14-4) if the nose gear door release lever is not latched properly.

DOWNLOCK PROVISIONS

In addition to downlock mechanisms within the nose gear drag stay and the main gear side stay, downlock (ground locking) pins are provided to prevent inadvertent gear retraction on the ground (Figures 14-1 and 14-3). The pins should be installed as soon as possible after landing and remain in position until immediately prior to engine start for the next flight. When not in use, the pins are stowed in a pouch in the vestibule or under the second step in the main entrance door.

CONTROLS AND INDICATORS

The landing gear is controlled by the LANDING GEAR selector lever on the center instrument panel (Figure 14-5). Gear position is indicated by three green and three red annunciators on the center instrument panel. Standby main gear downlock indication is provided by two green annunciators under a hinged cover on the right console. Additionally, a mechanical indicator pin located on the center pedestal extends as the nose gear locks down. A horn provides warning of abnormal conditions.

Controls

The LANDING GEAR selector lever is provided to control the normal retraction and extension of the landing gear. The lever operates in a two-gated slot in the center instrument panel and is connected to a hydraulic selector valve located under the flight compartment floor.

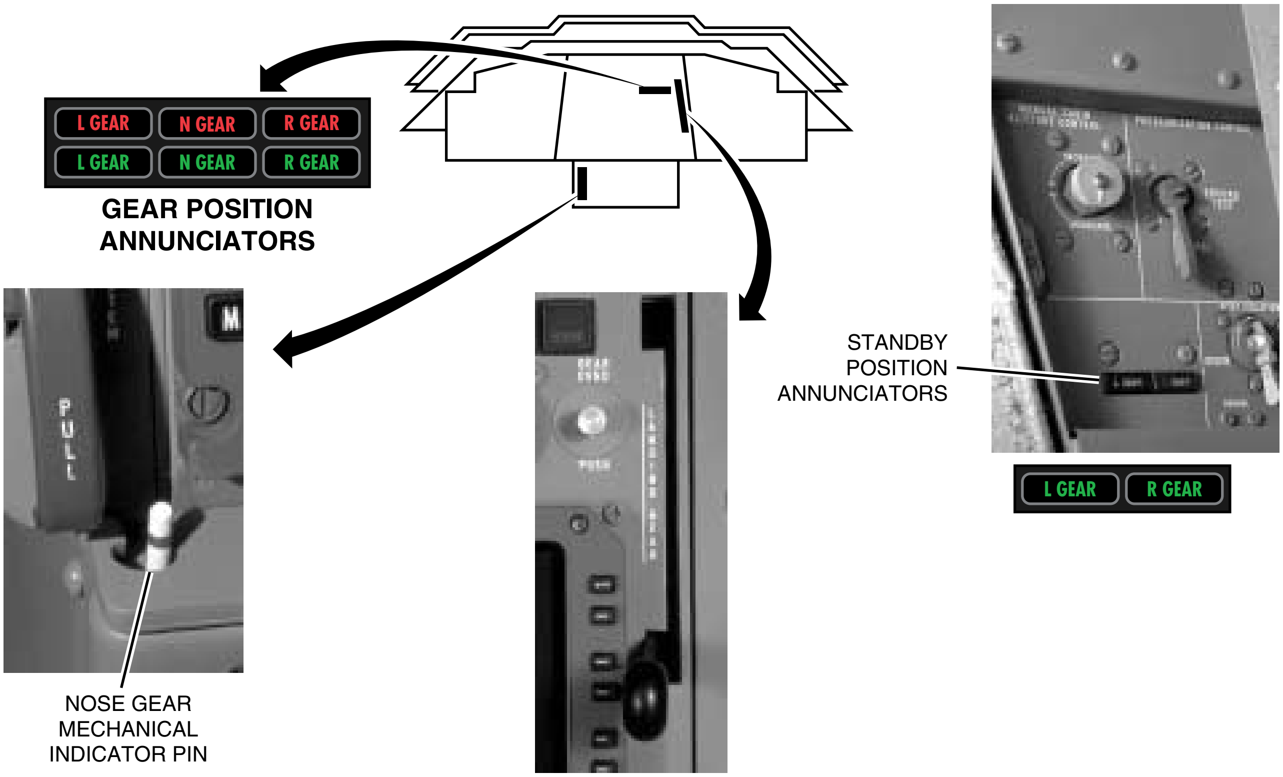

L GEAR
N GEAR
R GEAR
L GEAR
N GEAR
R GEAR
GEAR POSITION ANNUNCIATORS
PULL
NOSE GEAR MECHANICAL INDICATOR PIN
STANDBY POSITION ANNUNCIATORS
L GEAR
R GEAR

Figure 14-5. Landing Gear Controls and Position Indicators

When the airplane is on the ground, a solenoid-operated pawl engages to lock the LANDING GEAR selector lever in the down position. At takeoff, when the airplane weight comes off the wheels, the solenoid actuates to withdraw the pawl, allowing the lever to be moved away from the down position. The pawl reengages when the airplane lands.

Adjacent to the LANDING GEAR selector is the GEAR OVRD button. When the O/RIDE button is pressed, the landing gear lever lock is mechanically disengaged, allowing gear-up selection to be made (see Figure 14-6).

NOTE

> In an emergency, the landing gear can be retracted by pressing the GEAR OVRD button and selecting the LANDING GEAR lever up.

Indicators

Landing gear position is indicated by three green and three red annunciators (Figure 14-5). The green annunciators indicate gear down and locked. Illumination of red annunciators indicates gear in transit, gear unlocked, or gear down and locked with the gear selector lever out of the down position. In the last situation, all six lights are illuminated. With both main gears retracted, the red annunciators go out when the inboard doors close. With all gear up and locked, all six annunciators are out. In addition, the two green standby main gear annunciators on the right console, controlled by a separate circuit, come on as a backup indication when the main gears are down and locked. A mechanically operated nose gear indicator pin, when extended, indicates nose gear down and locked.

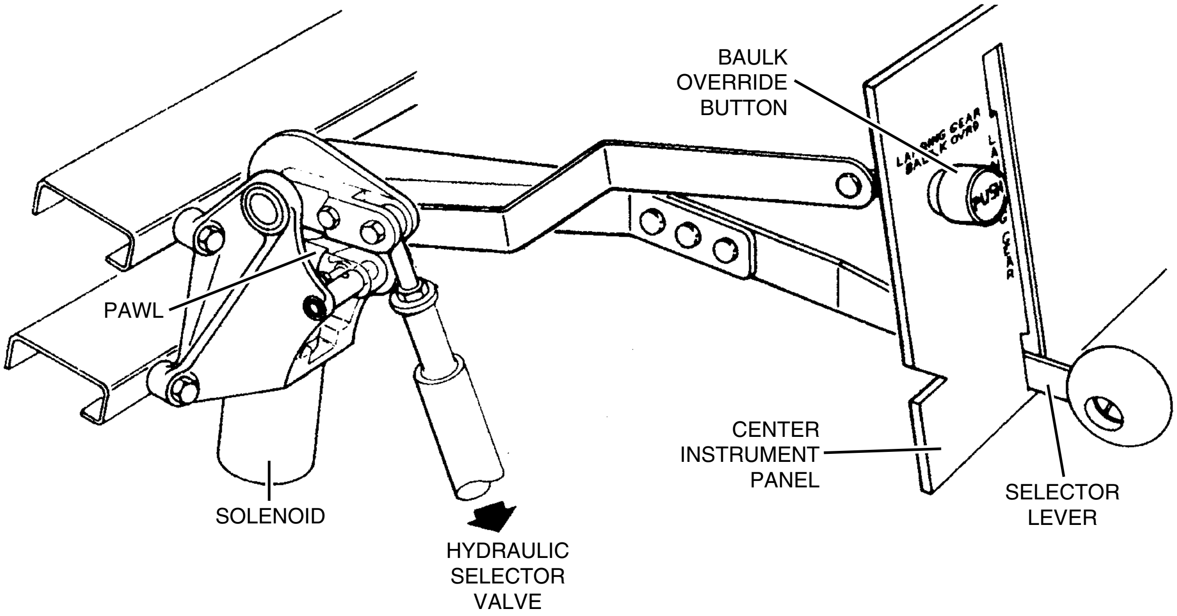


Figure 14-6. LANDING GEAR Selector Lever Locking Device

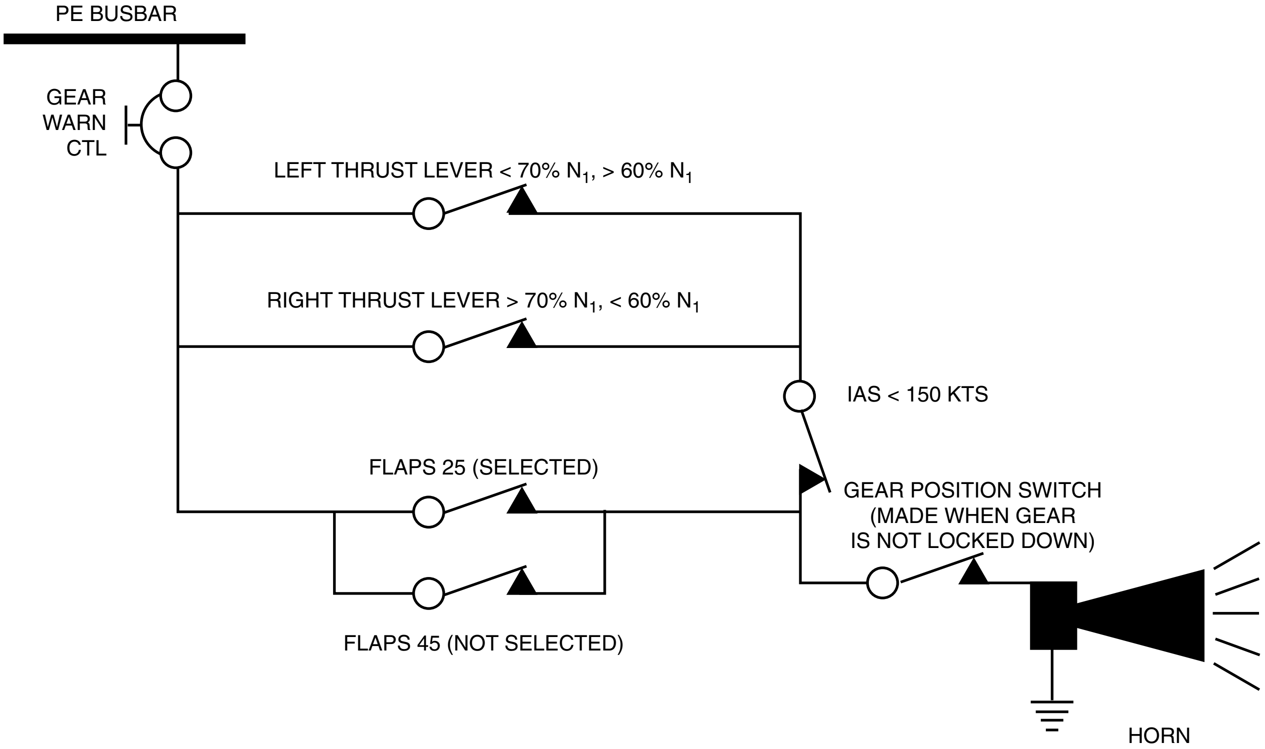

PE BUSBAR
GEAR WARN CTL
LEFT THRUST LEVER < 70% N1, > 60% N1
RIGHT THRUST LEVER > 70% N1, < 60% N1
IAS < 150 KTS
FLAPS 25 (SELECTED)
GEAR POSITION SWITCH (MADE WHEN GEAR IS NOT LOCKED DOWN)
FLAPS 45 (NOT SELECTED)
HORN

Figure 14-7. Gear Warning Horn Circuit Schematic

Warning System

A warning horn sounds if the landing gear is not down and locked and either of the following conditions exist:

- The flaps are extended to the approach or landing position.
- Either throttle is closed to obtain between 60 and 70% N_1 rpm (nominal) with IAS below 150 kt. The audible warning is repeated if the second throttle is closed after a previous warning being cancelled and not reinstated.

NOTE

The warning horn also sounds if the landing gear is down and locked, but the LANDING GEAR selector lever is out of the down position, and the START PWR switch is selected to the PWR ON position.

OPERATION

General

The main gear side stays and the nose gear drag brace (Figures 14-1 and 14-3) incorporate integral locking mechanisms which lock the gear in both the extended and retracted positions. Initial movement of the actuator releases the locks; they are engaged by final movement of the actuators. Main gear inboard doors are sequenced to the closed position when the gear is in either the extended or the retracted position. Nose gear doors are mechanically operated by gear movement. The forward doors are closed with the gear extended; the aft door remains open.

Normal Retraction

Placing the LANDING GEAR lever up mechanically positions the gear selector valve to direct main system pressure to open the main gear inboard doors and disengage the downlocks. The gear retracts, the final movement of the actuators engaging the uplocks. The main gear inboard doors hydraulically close and lock; the nose gear doors are mechanically closed.

Normal Extension

Placing the LANDING GEAR lever down reroutes flow through the selector valve, directing pressure to unlock the main gear and inboard doors and to unlock the nose gear. The gear extends and the last movement of the actuators locks the main gear stay and the nose gear drag brace. The main gear inboard doors close, the nose gear doors close, and the aft door remains open.

Emergency Extension

The system is prepared for use by firmly pulling a red AUX HYD SYSTEM lever. Pulling the handle:

- Closes an on-off valve (normally open during main system operation to relieve fluid seepage)
- Operates a dump valve to isolate any main system pressure, and to interconnect the normal up and down pressure lines of the main and nose gear actuators with the main system return line
- Moves a balk clear of the handpump operating handle socket

The AUX HYD SYSTEM lever is held in the selected position by a spring.

An auxiliary hydraulic handpump is provided for in-flight use only. The handpump operating handle is removable and is normally stowed. Inserting the handle into the handpump socket permits pump operation.

Operation of the handpump draws fluid from the auxiliary tank. The pressurized fluid is delivered through a check valve and shuttle valves to the down side of the main and nose gear actuators, and to the flap control unit.

NOTE

Shortly after emergency extension begins, handpump resistance decreases (when the landing gear uplocks release) and the AUX HYD LO LEVEL annunciator illuminates (all colored annunciators are shown in Appendix B). Continued handpump operation locks the gear down and, after appropriate selection, achieves the required flap setting.

Following an emergency extension of the landing gear, the AUX HYD SYSTEM lever must not be reset. Proper positioning of the lever and the auxiliary handpump socket must be accomplished prior to flight to assure gear retraction.

Emergency extension is not predicated on the LANDING GEAR lever being in the down position.

NOTE

With the main system functioning normally, pulling the AUX HYD SYSTEM lever has no effect on flaps operation. If an emergency extension is selected to demonstrate the system, the flaps will be operated by main system pressure.

CAUTION

When operating the auxiliary system to lower the landing gear, the hand pump action must be continued, after three greens are achieved, until positive resistance is felt to make sure the landing gear is down and locked.

NOSEWHEEL STEERING

GENERAL

The manually controlled, hydraulically actuated nosewheel steering system provides deflection 45° right and left of center. A hydraulic actuator (steering jack) provides movement through mechanical linkage to steer the nosewheels.

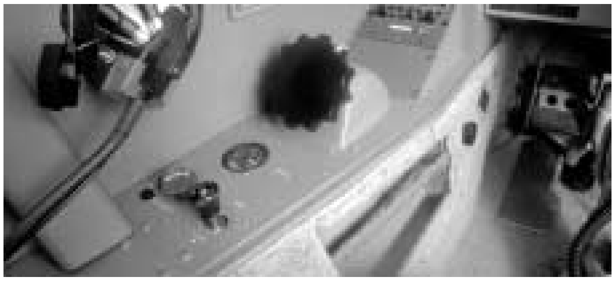
OPERATION

Power for the steering system is provided by the main hydraulic system. Rotation of the handwheel on the pilot's console (Figure 14-8) moves the selector valve of the steering actuator which directs pressure to turn the nosewheel appropriately (Figure 14-9). When the selected position is reached, the nosewheel steering angle is maintained. A relief valve, fitted into the actuator hydraulic return line, reduces the possibility of nosewheel shimmy.

When the nose gear unlocks prior to retraction, a shutoff valve is mechanically closed, rendering the steering system inoperative.

WARNING

> The steering handwheel must be free from obstruction during gear lowering, otherwise an unwanted steering command will be given which may result in the aircraft veering off the runway immediately on touchdown
>
> The reason is, the geometry of the linkage from the handwheel to the selector valve causes the handwheel to rotate while the gear is lowering or retracting. The handwheel must be free to turn and to find its neutral position again prior to the gear locking down.



PILOT'S CONSOLE

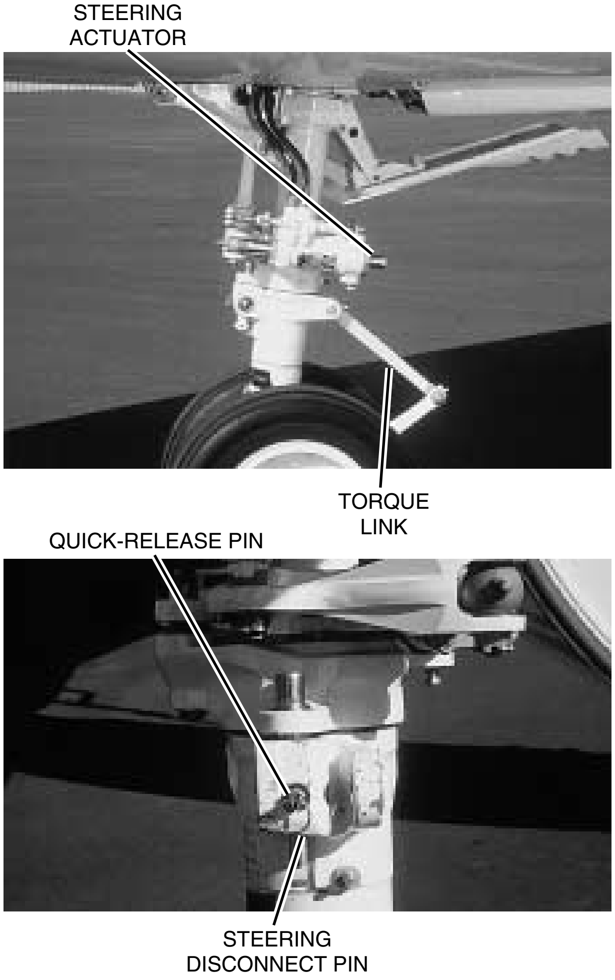


Figure 14-8. Nosewheel Steering System

The quick-release and steering disconnect pins must be removed prior to towing. Install the steering disconnect pin, retaining it with the quick-release pin, for all towing operations.

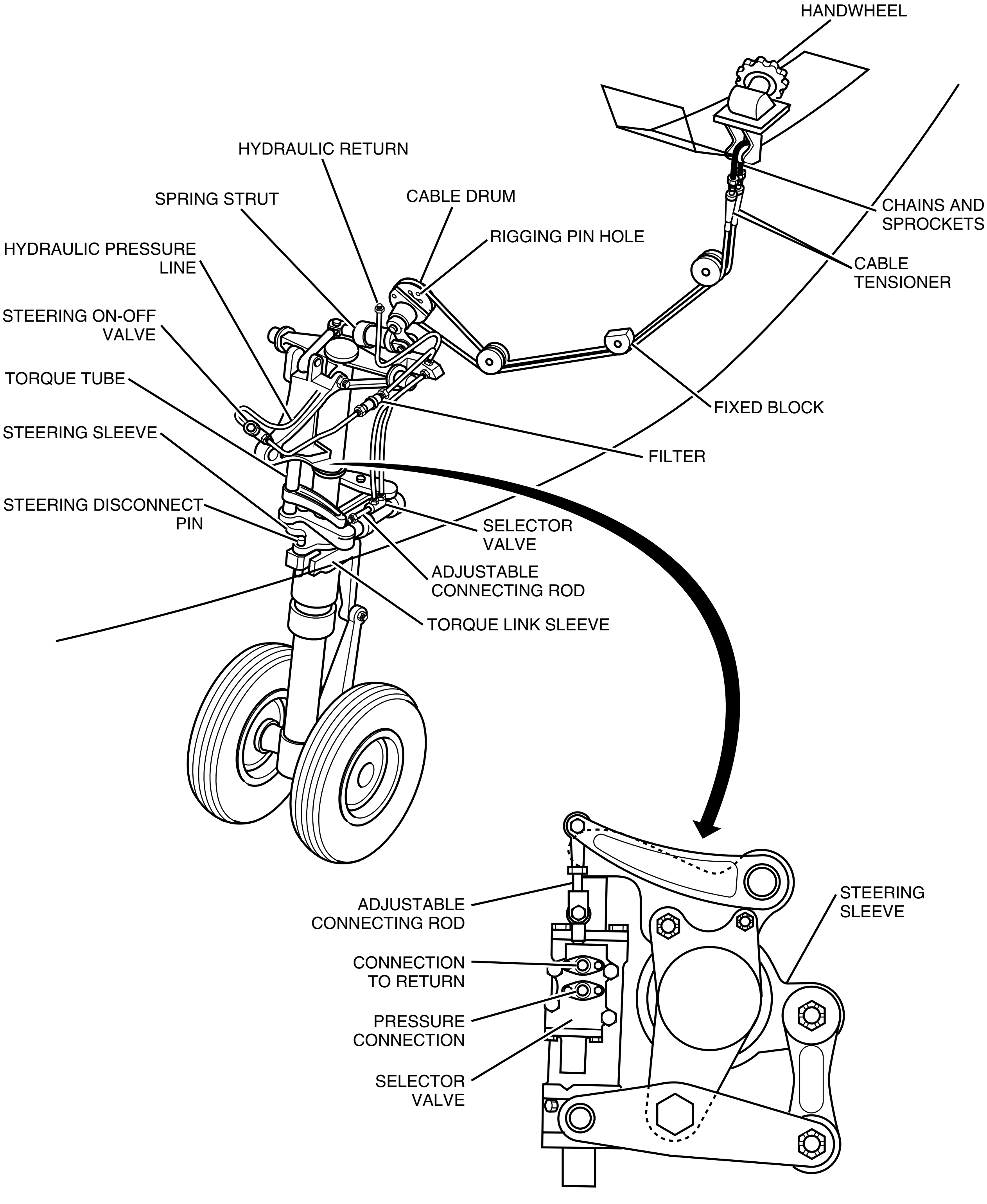


Figure 14-9. Nosewheel Steering System Diagram

CAUTION

The steering disconnect and quick-release pins must be installed prior to taxiing.

CAUTION

Noswheel steering is not available when the main hydraulic pressure is less than 2,300 psi.

The auxiliary hydraulic system cannot power the steering circuits.

BRAKES

GENERAL

The main hydraulic system provides pressure to actuate the caliper disc-type, self-adjusting brake assemblies, one on each main gear wheel. Master cylinders, operated by mechanical linkage from the tops of the rudder pedals, provide control pressure for brake operation. Selection of normal, emergency, or parking brake is made with the WHEEL BRAKE lever on the center pedestal. Antiskid protection is provided during normal braking only.

OPERATION

Normal

Normal braking is available with the WHEEL BRAKE lever in the NORMAL (full forward) position. Main hydraulic system pressure backed by the main accumulator is reduced and supplied to the brake control valves (Figure 14-10). Pressure applied to the toe brake pedals by either the pilot or copilot provides master cylinder pressure to position the brake control valves.

Braking pressure, in proportion to pedal force, is directed through modulators to the brakes.

Wheel-driven antiskid (Maxaret) units provide constant antiskid protection by detecting excessive wheel deceleration and releasing braking pressure as required with wheel rotation.

Pressure in the brake lines being applied to the brakes is indicated on the BRAKES scale of the DC-powered hydraulic pressure indicator. Relieving toe pressure on the toe brake pedals relieves master cylinder pressure on the brake control valve, venting braking pressure to the return system and releasing the brakes.

Main Hydraulic System Failure

General

When main hydraulic system failure occurs, a pressure-maintaining valve closes to trap main system accumulator pressure for braking. Normal braking with Maxaret control is available for landing under average conditions. Wet runway conditions will decrease the applications available.

CAUTION

After landing under the above conditions, the aircraft should not be taxied.

Selection of emergency brakes should be delayed until main system accumulator pressure is exhausted, since Maxaret control is not available with emergency brakes. With a fully charged emergency brake accumulator, approximately 18 brake applications are available.

CAUTION

During the landing roll, the brakes should be applied gently. The aircraft should not be taxied.

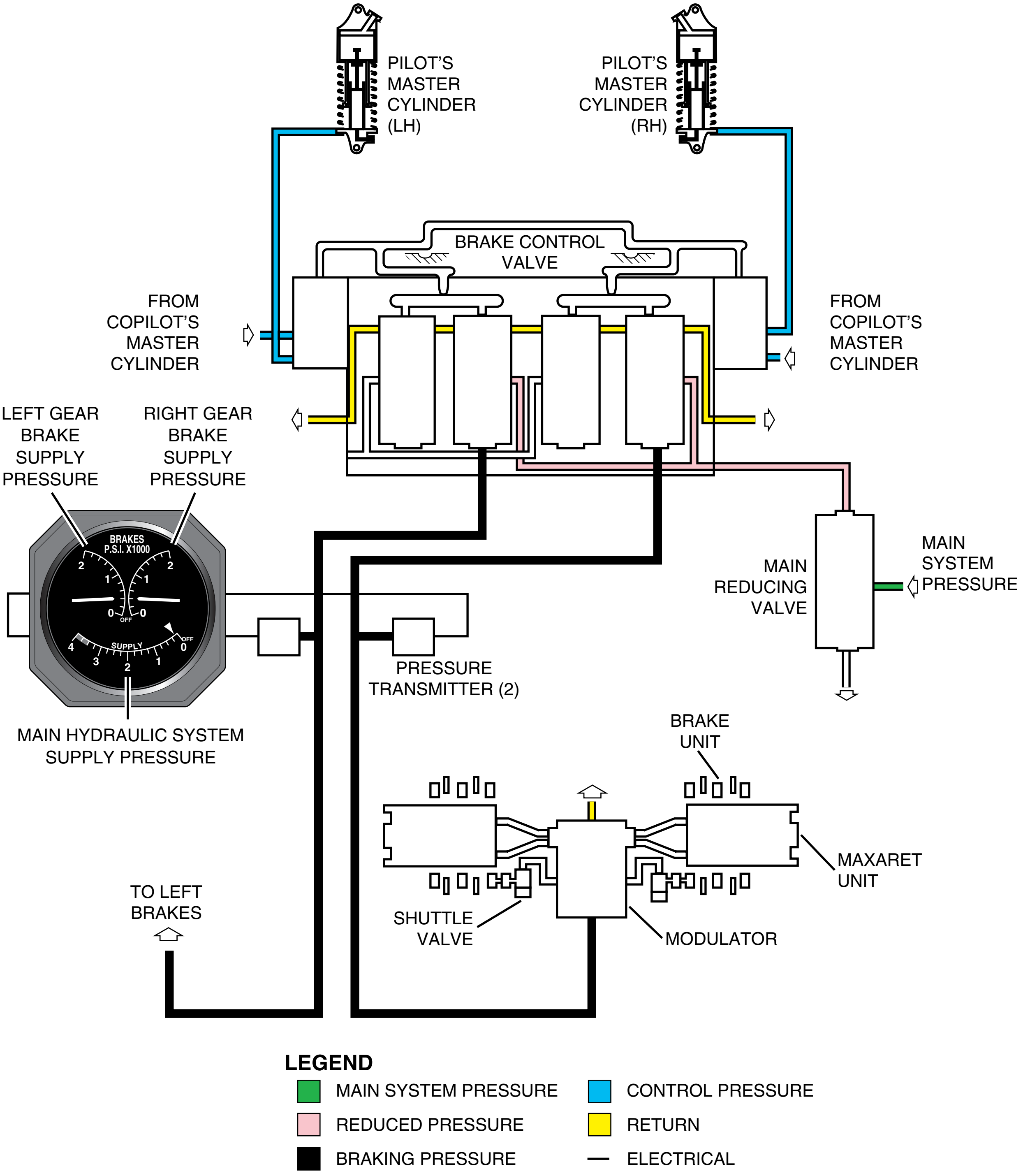


Figure 14-10. Normal Brake System

Operation

Emergency brake operation is selected by moving the WHEEL BRAKE lever (Figure 14-11) aft to the EMERGY position (first notch marked in red), mechanically opening the emergency brake reducing valve (Figure 14-12). Emergency brake accumulator pressure is reduced and applied to the brake control valves. As in normal operation, pressure on the toe brake pedals produces master cylinder control pressure to operate the brake control valves. Braking pressure, in proportion to pedal force, is directed through shuttle valves to the brake assemblies, bypassing the modulators and anti-skid units. The white L and R WHEEL BRAKE EMERG annunciators come on as braking pressure in the respective gear brake line reaches 1,000 psi.

Relieving toe pressure on the pedals releases the brakes in the manner described under Normal Operation. As the brakes release, the L and R WHEEL BRAKE EMERG lights go out.

Reduction of emergency brake accumulator pressure below 2,250 psi is indicated by illumination of the EMRG BRK LO PRESS annunciator.

Parking

With pressure available from both the main and emergency brake accumulators, both brake systems are available for setting the parking brake. Movement of the WHEEL BRAKE lever aft into the PARK BRAKE range (Figure 14-13) progressively applies the brakes. At 1,000 psi applied pressure, the WHEEL BRAKE EMERG annunciators illuminate white. The brake pressure applied to the left and right brake units is also displayed on the combined indicator to show normal system pressure. Full parking pressure is applied when the WHEEL BRAKE lever is in the last notch. Pressure supplied by the emergency brake accumulator and the main accumulator is maintained at the brakes by a lever that actuates the brake control valve. A spring strut, initially loaded by aft movement of the WHEEL BRAKE lever, resets the control valve when the lever is released to NORMAL or EMERGY.

ELECTRICAL POWER SOURCES

PE bus bar

Normal landing gear position annunciators (6)

Warning horn control

Warning horn

WHEEL BRAKE EMERG annunciators L and R

PS1 bus bar

Landing gear lever lock solenoid

Normal BRAKES applied pressure indicator

PS2 bus bar

Standby landing gear downlock annunciators (2)

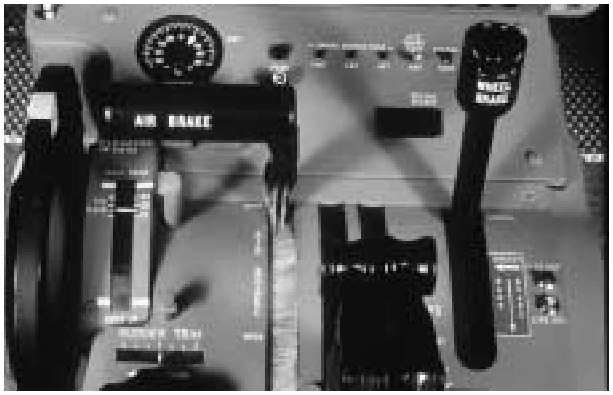
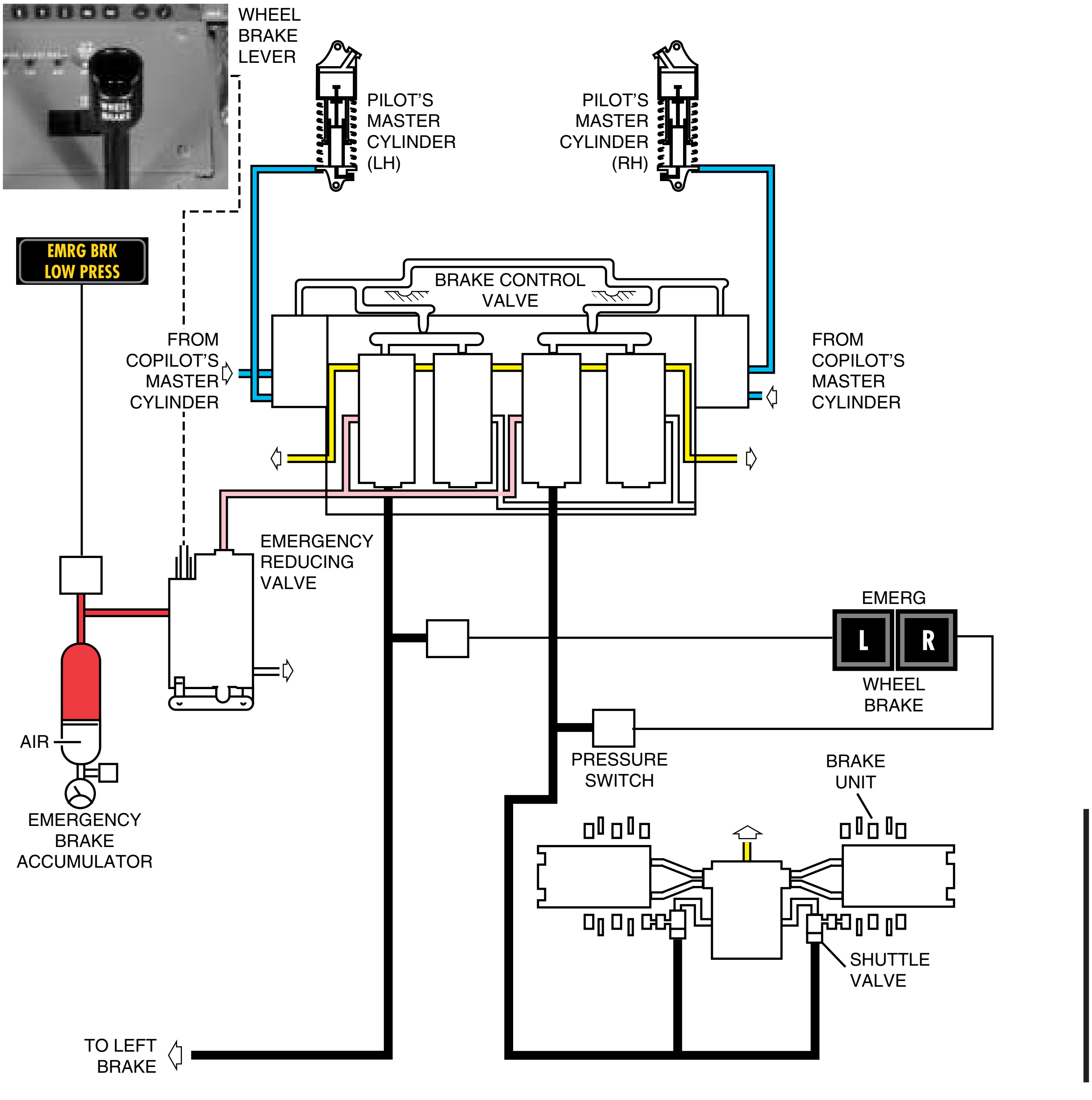


Figure 14-11. Wheel Brake Selector Lever



LEGEND

🟥 EMERGENCY ACUMULATOR PRESSURE

🟥 REDUCED PRESSURE

■ BRAKING PRESSURE

🟦 CONTROL PRESSURE

🟨 RETURN

-- MECHANICAL

— ELECTRICAL

Figure 14-12. Emergency Brake System

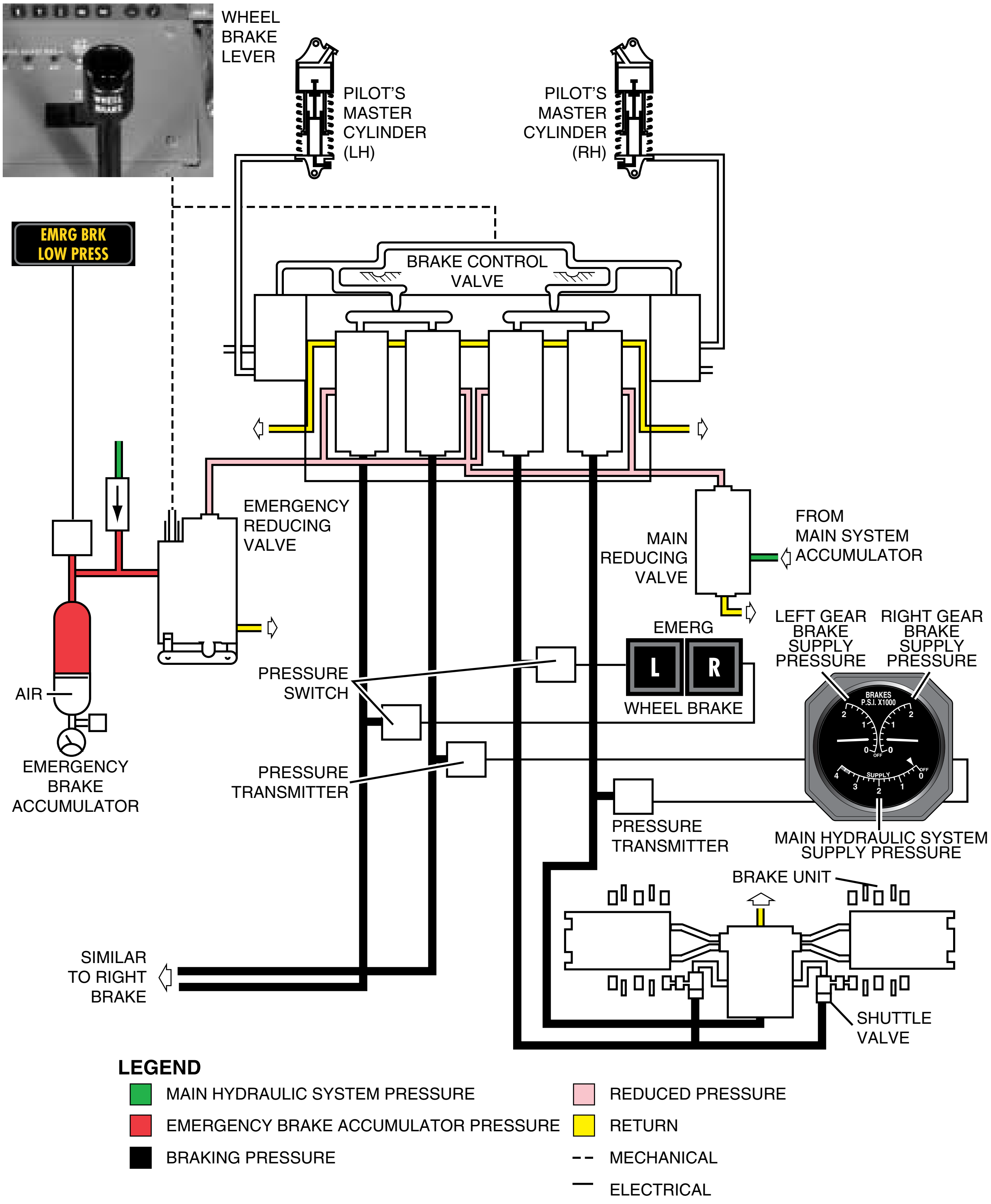


Figure 14-13. Parking Brake System

LIMITATIONS

LANDING GEAR

Landing Gear Operating Speed (V_{LO} and V_{LE})

The maximum permissible airspeed for extending or retracting the landing gear and with the gear extended is 220 knots IAS.

BRAKES

Brake Cooling Periods

The normal wheel brake system embodies Maxaret anti-skid units which automatically reduce the brake pressure should a wheel tend to skid. As the Maxaret unit does not operate until the wheel is revolving, the brakes must not be applied before touchdown. It should be noted that the emergency braking system bypasses the anti-skid units and therefore care should be exercised when using this system.

If any of the wheels' fusible plugs blow, the brakes must be inspected and certified serviceable before the next takeoff.

The brakes are of adequate capacity to bring the aircraft to a stop under all circumstances (including a rejected takeoff from V_1) provided that the procedures stated below have been carried out.

Cooling Times

After the aircraft has made a normal landing or a stop from a rejected takeoff, a waiting period should be established to ensure that the brakes are both sufficiently cool and in a serviceable condition for a further rejected takeoff (critical case).

After Normal Landing

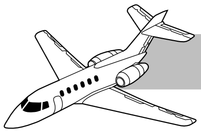
The required waiting period from completion of taxi-in from landing to before start of taxi-out for takeoff is five minutes, except when the takeoff weight exceeds the values

Table 14-1. TAKEOFF WEIGHTS VS BRAKE COOLING PERIODS

AIRFIELD ALTITUDE (FT)	TEMPERATURE (° C)						
	-10	0	+10	+20	+30	+40	+50
	TAKEOFF WEIGHT (LBS)						
10,000	22,300	21,900	21,500	21,100	19,500		
9,000	22,800	22,400	21,900	21,600	20,300		
8,000	23,300	22,800	22,400	22,000	21,100		
7,000	23,800	23,300	22,900	22,400	21,900		
6,000	24,300	23,800	23,300	22,900	22,500	20,900	
5,000	24,700	24,200	23,800	23,300	22,900	21,600	
4,000	25,200	24,700	24,300	23,800	23,400	22,400	
3,000	25,700	25,200	24,700	24,300	23,800	23,100	
2,000	26,300	25,700	25,200	24,700	24,300	23,900	
1,000	26,800	26,200	25,700	25,200	24,700	24,300	23,300
Sea Level	27,300	26,700	26,200	25,700	25,200	24,800	24,100

NOTE:

1. In 1-5 knots tailwind subtract 500 lb.
2. In 6-10 knots tailwind subtract 1,500 lb.
3. If the downhill slope exceeds 1/2%, subtract 500 lb.



given in the table below. When the weight exceeds these values, a period of 30 minutes must be allowed.

The table is based on still air and a downhill slope not exceeding 1/2%. Corrections for more adverse conditions are given in the NOTES.

After Rejected Takeoff

Required period from completion of taxi-in following a rejected takeoff from a speed of 90 knots or less, to before start of taxi-out for takeoff.

After a single
rejected takeoff .... 25 Minutes

After two or more successive
rejected takeoffs............ 45 Minutes

If the rejected takeoff is made from a speed greater than 90 knots IAS the brakes must be inspected and certified to be serviceable before the next takeoff.

CREW TRAINING

The waiting times obtained by the table on the previous page relate to normal operations in which the brakes and tires have time to cool between landings. If repeated braked landings are made for crew training or any other reason, the brakes and tires may not have time to cool between runs and their temperatures may rise to an undesirable level. The following restrictions should therefore be observed.

Heavy braking should not be used more than necessary for the purpose of the exercise, and the landing gear should be extended as long as possible—never less than five minutes in each circuit.

Landings with light braking may be repeated at intervals of not less than 15 minutes. After a landing with heavy braking, one or more touch and go circuits should be done, and 30 minutes should elapse before the next braked landing.

QUESTIONS

1. Inadvertent landing gear retraction on the ground is prevented by:
   A. A solenoid-operated device engaging the selector lever
   B. The squat switch opening the circuit to the solenoid-operated landing gear selector valve
   C. Inability of the hydraulic system to retract the gear while on the ground
   D. None of the above

2. When all landing gear are up and locked, position indications are:
   A. Three red, three green annunciators on
   B. Three green, no red annunciators on
   C. No green, no red annunciators on
   D. None of the above

3. The main landing gear inboard doors are closed:
   A. Only when the gear is retracted
   B. Only when the gear is extended
   C. When the gear is in either the extended or the retracted position
   D. Only when the gear has been extended with the auxiliary system

4. The landing gear warning horn will sound when:
   A. The gear is not locked down with flaps selected to 25°.
   B. The gear is not locked down with either power lever between 60% and 70% N_1, and airspeed is below 150 KIAS.
   C. The gear is down and locked, the LANDING GEAR lever is not selected down, and start power is selected.
   D. All the above

5. An indication of landing gear down and locked (3 green lights) accompanied by 3 red lights indicates:
   A. A malfunction
   B. The LANDING GEAR lever is not in the down position and the landing gear is down and locked.
   C. The gear is in transit.
   D. B and C are correct.

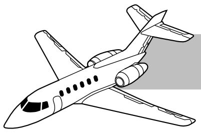
6. If the nose gear does not lock down during extension, the indication is:
   A. The red N GEAR light on the annunciator panel is on, and the mechanical indicator on the pedestal is not extended.
   B. The red N GEAR light on the right console is on.
   C. The mechanical nose gear indicator on the left side of the pedestal is extended.
   D. All three red indicator lights on the annunciator panel are on.

7. Nosewheel steering with the gear retracted is prevented by:
   A. A mechanically actuated hydraulic shutoff valve
   B. The nose gear strut centering mechanism
   C. A mechanical lockout that prevents rotation of the handwheel on the left console
   D. None of the above

8. The EMRG BRK LO PRESS annunciator comes on:
   A. When emergency brake accumulator pressure drops to 0 psi
   B. When emergency brake accumulator pressure drops to 2,250 psi
   C. When the emergency brake system is put into operation
   D. When less than 1,000 psi is being applied to the brakes

9. Antiskid protection is available:
   A. When braking is from the normal system
   B. When braking is from the emergency system
   C. With a switch on the pedestal during any braking
   D. With a switch on the pedestal during normal braking

10. In order to select emergency brakes, the WHEEL BRAKE lever must be positioned:
    A. Full forward
    B. Full aft
    C. First aft notch marked in red
    D. As far aft as possible after depressing button



CHAPTER 15
FLIGHT CONTROLS

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CHAPTER 15
FLIGHT CONTROLS

INTRODUCTION

The primary flight controls of the Hawker 800 XP airplane are manually actuated. Trim is mechanical, with electrical elevator trim also provided.

Secondary flight controls consist of flaps and airbrakes. A stall warning and identification system warns of impending stalls by shaking or pushing the control columns.

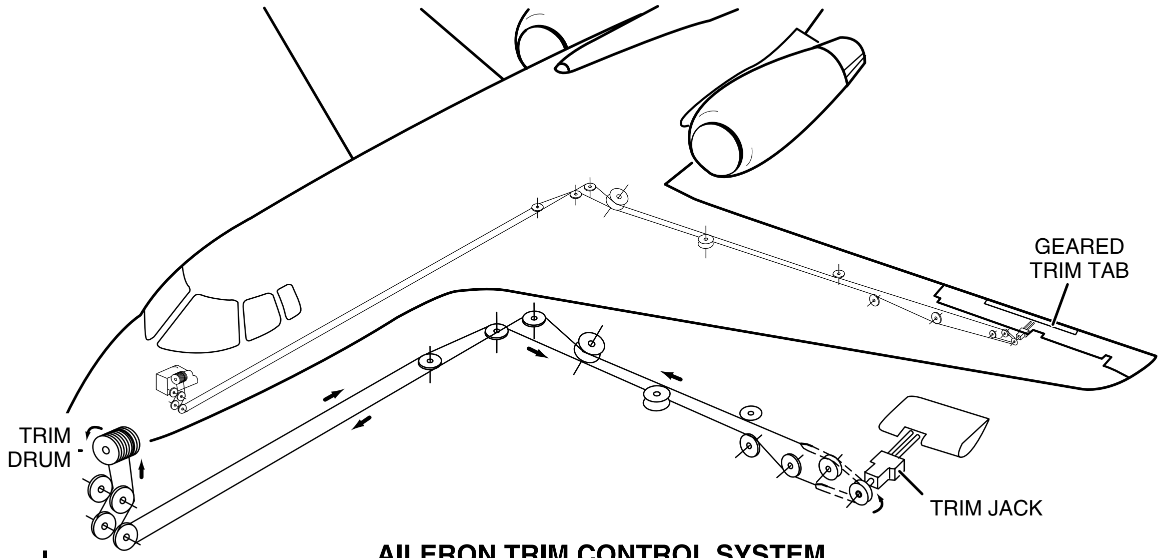
GENERAL

Primary controls (ailerons, elevator and rudder), from dual ram's-horn type control columns and adjustable rudder pedals, are operated manually by cable-activated circuits. These control surfaces are mass and aerodynamically balanced.

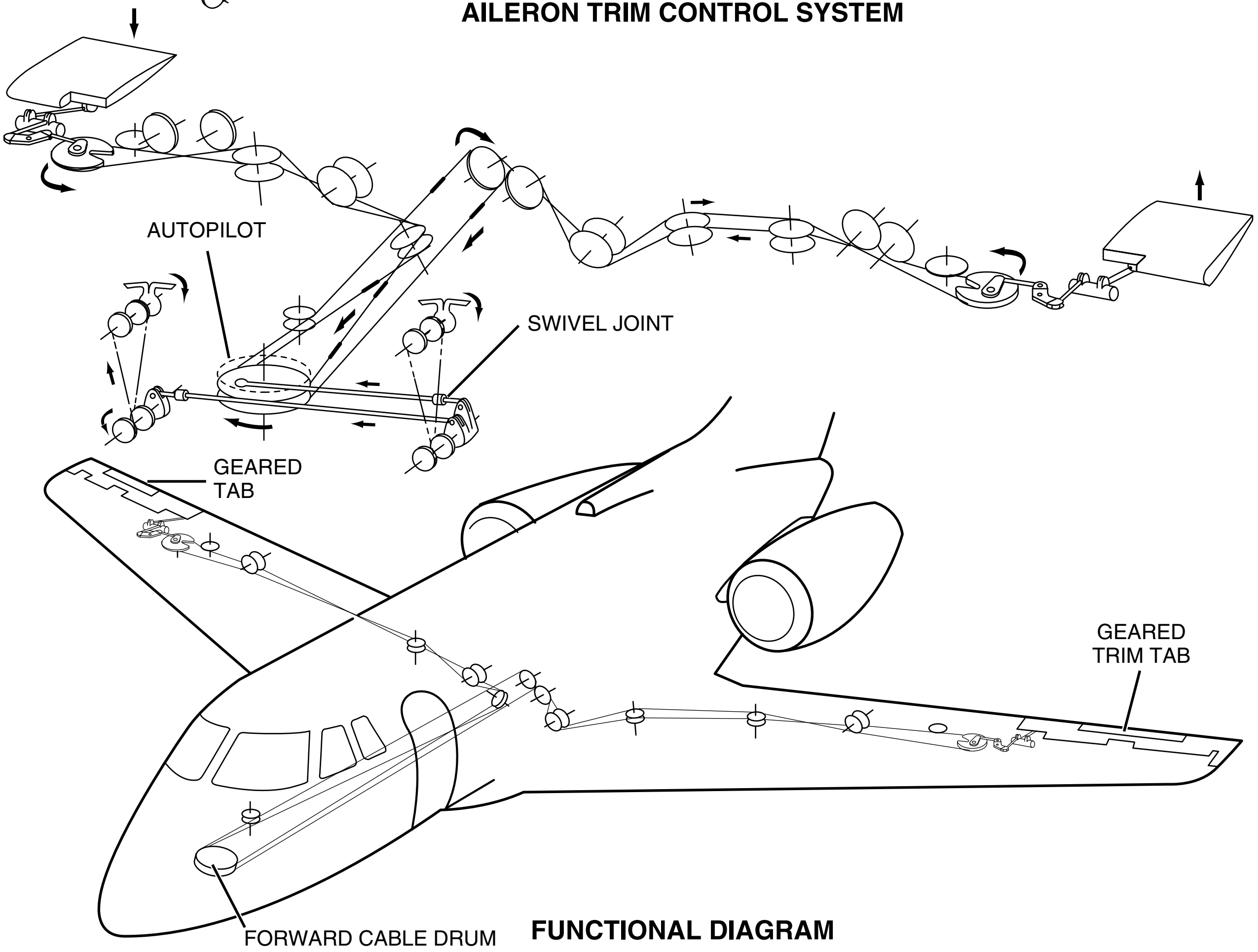
Trim tabs, operated from handwheels on the control pedestal, are fitted to the rudder, elevator and left aileron. The elevator trim tab may also be servomotor-operated by the pilot (electric trim), the autopilot, or the Mach trim system.

Airbrakes are powered by the main hydraulic system. AIRBRAKE selector lever labeled positions are SHUT and OPEN, with an infinite selection of intermediate positions. A third labeled position, DUMP, is available after lifting the selector lever through a baulk at the OPEN position.

On the wing trailing edge, hydraulically powered flaps are installed. Labeled selected positions are: 0° (UP), 15°, 25°, and 45° (FULLY DOWN). A lift-dump position (75°) is available via the airbrake DUMP selection. Emergency lowering is available via an auxiliary hydraulic system. A flap



AILERON TRIM CONTROL SYSTEM



FUNCTIONAL DIAGRAM

Figure 15-1. Aileron System

position indicator is provided in the center instrument panel.

A rudder bias system is powered by engine bleed air. This system automatically applies opposite rudder to counteract the effect of asymmetric thrust resulting from a single-engine failure.

A stall warning (stick shake) and stall identification system (stick push) is provided.

Internal gust locks are provided to lock the control surfaces when the aircraft is parked or moored. The gust locks are rated for gusts of up to 80 knots.

PRIMARY CONTROLS

AILERON CONTROL SYSTEM

Movement of the ram's-horn type handwheel on either control column operates the ailerons. Each handwheel operates a lever at the bottom of the column through chains and cables. A connecting rod interconnects the levers of each column. The left column lever is also connected to a pulley drum. From the pulley drum a combination of cables and tie-rods operate a differential pulley in each wing. This pulley is connected to the aileron by the levers and links (Figure 15-1).

Each aileron control surface is fitted with a mass balance weight to prevent flutter and a horn balance to reduce the aerodynamic loads felt by the pilot.

Primary stops are located at the left and right aileron inboard hinges to limit the upward range of movement. Secondary stops are located at the bottom of each control column.

Aileron Geared and Trim Tabs

Each aileron has a servo-action geared tab mounted on its trailing edge (Figure 15-2). The tab is connected to the aileron by twin connecting rods, and moves in the opposite sense to the aileron lightening the aerodynamic loads on the pilot's handwheel.

The connecting rods for the left aileron tabs are connected to a manually operated trim actuator. The trim actuator is connected by a cable system to a trim handwheel mounted on the center pedestal in the flight compartment.

Aileron Out-of-Trim Warning

A warning of the aileron being out of trim for takeoff is given by the ELEV/AIL TRIM annunciator illuminating (on the main MWS panel) if the following conditions apply:

- Aircraft weight is on wheels.
- Both engine thrust levers are advanced to more than 40–60% N_1 power.

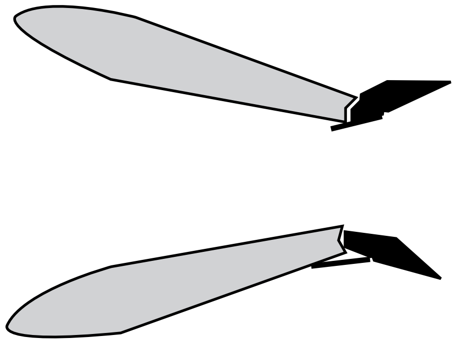


Figure 15-2. Aileron Geared Tab Servo-Action

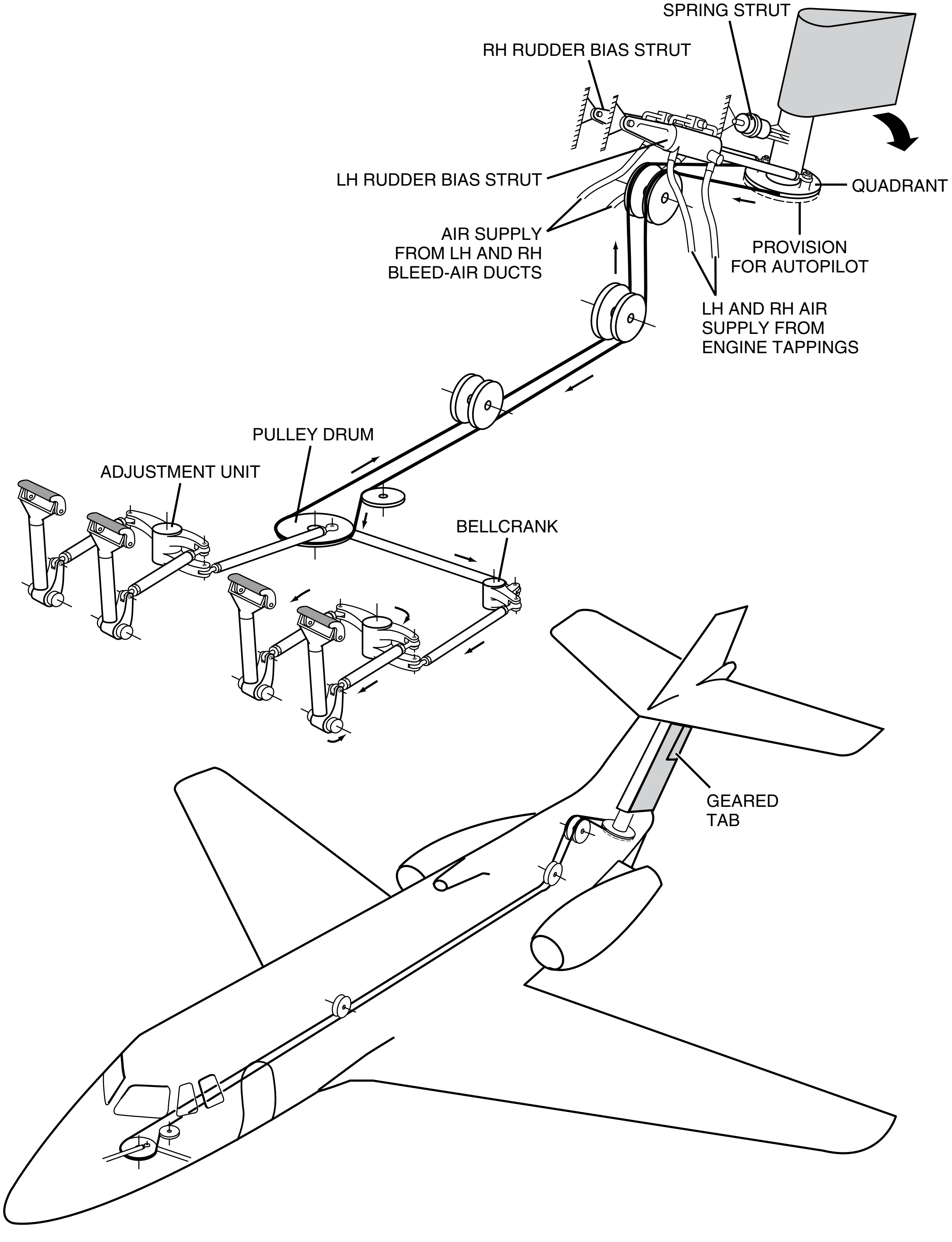


Figure 15-3. Rudder System

RUDDER CONTROL SYSTEM

The rudder is hinged from the rear of the vertical stabilizer and extends from the rear cone of the fuselage to just below the undersurface of the horizontal stabilizer (Figure 15-3).

In the flight compartment, each pilot is provided with a pair of rudder pedals which incorporate adjustment for reach.

Movement of the rudder pedals is transmitted by connecting rods and a bellcrank to a pulley drum under the flight compartment floor. Cables transmit the movement of the pulley drum to a quadrant fitted to the bottom of the rudder torque tube.

A spring strut connected to the lower portion of the rudder torque tube provides a centering force at the extreme travel. An autopilot servomotor is also connected to the base of the torque tube.

Primary stops are located at the rudder lower hinge to determine the range of movement in both directions. Secondary stops are located at each rudder pedal assembly.

Flow separation triggers have been installed on both sides of the rudder to reduce the possibility of too much rudder application during rudder bias operation (Figure 15-3A).

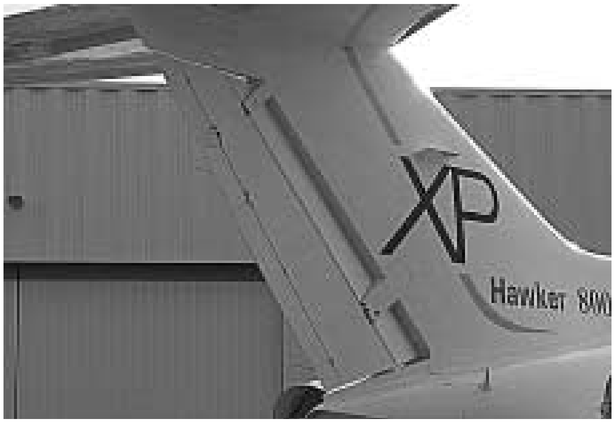


Figure 15-3A. Rudder

Rudder Geared and Trim Tabs

Two separate tabs are hinged at the trailing edge of the rudder. The upper tab is linked to the rudder by two connecting rods, the lower tab is linked by rods to a screw trim actuator unit. Both tabs provide an antiservo action. The position of the lower tab may be controlled manually by the rudder trim handwheel in the flight compartment (Figure 15-4).

Rudder Bias System

Two air-powered struts are connected between the fuselage and the rudder torque tube quadrant providing an automatic application of rudder bias to counteract asymmetric thrust caused by failure or malfunction of one engine (Figure 15-5).

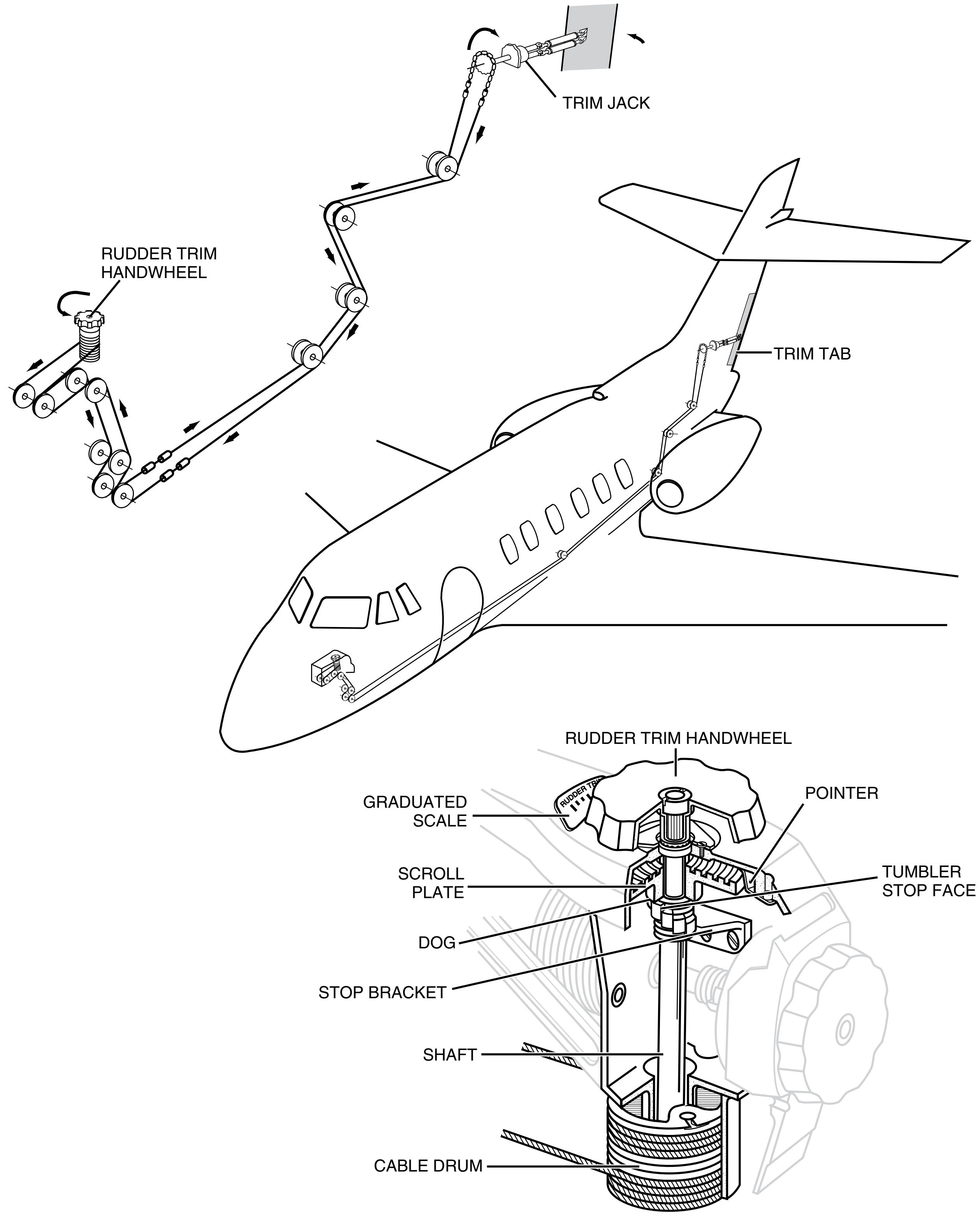
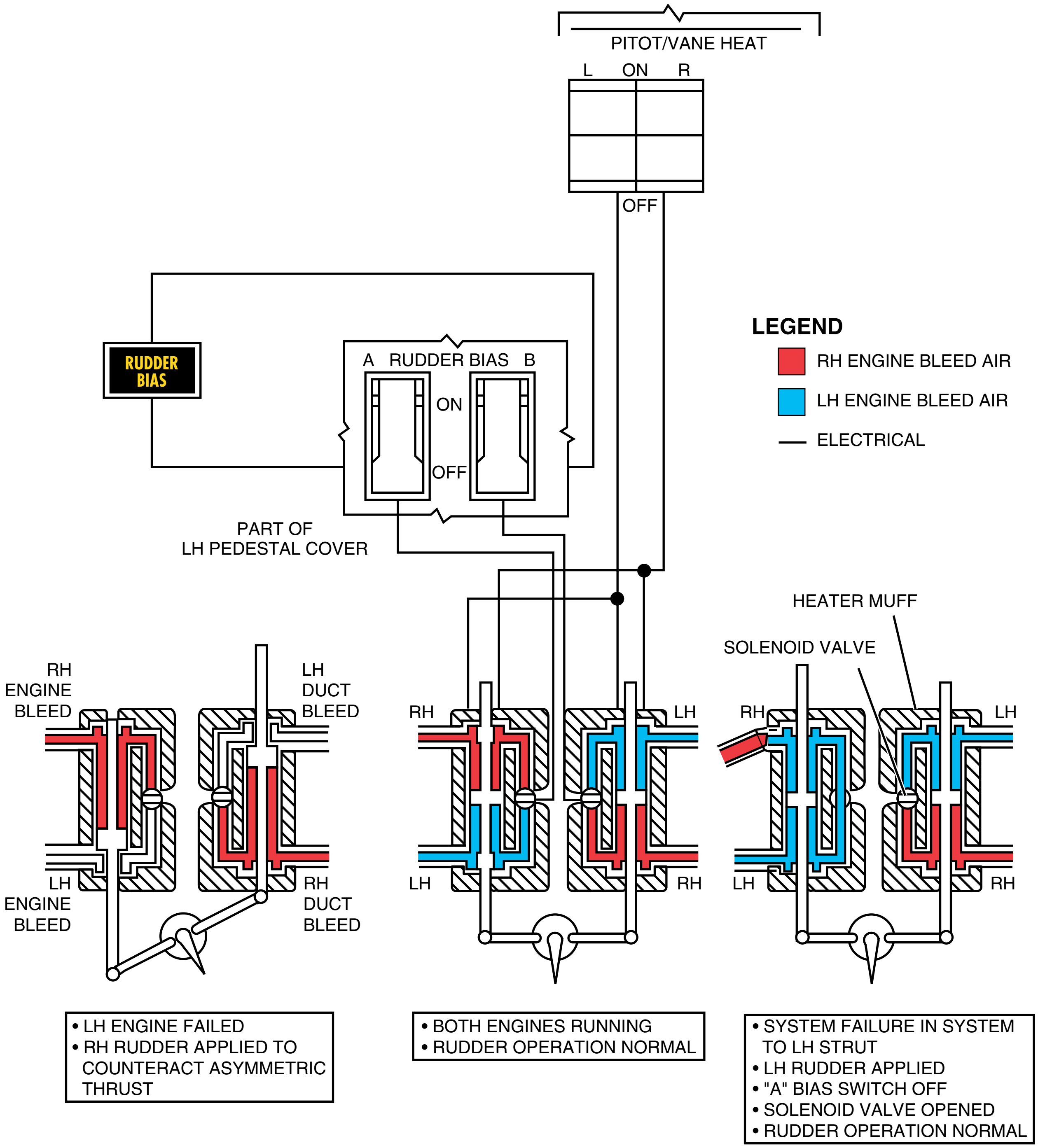


Figure 15-4. Rudder Trim System



- LH ENGINE FAILED
- RH RUDDER APPLIED TO COUNTERACT ASYMMETRIC THRUST

- BOTH ENGINES RUNNING
- RUDDER OPERATION NORMAL

- SYSTEM FAILURE IN SYSTEM TO LH STRUT
- LH RUDDER APPLIED
- "A" BIAS SWITCH OFF
- SOLENOID VALVE OPENED
- RUDDER OPERATION NORMAL

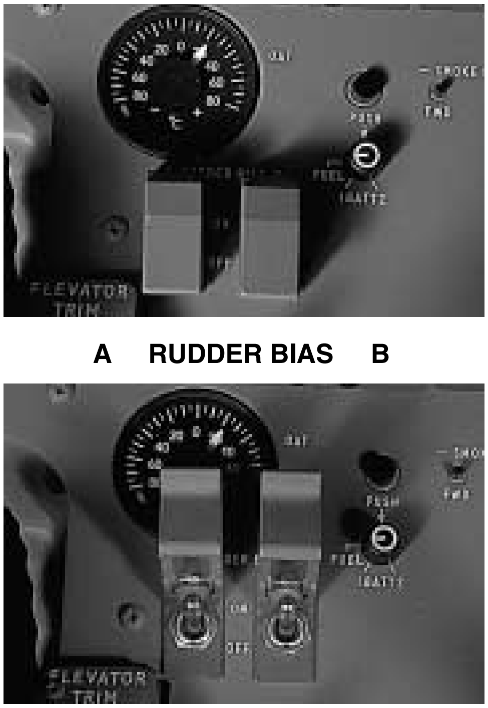
Figure 15-5. Rudder Bias System Design

The engine bleed-air system is interconnected to the struts in such a manner that each engine supplies air to opposing sides of the strut pistons.

A solenoid valve is fitted in each strut. The solenoid valves are normally in the closed position when the RUDDER BIAS selector switches (A and B) are set to their ON positions. The switches are located on the forward extension of the center control pedestal (Figure 15-6).

Accidental operation of the switches is prevented by a guard on each switch. An amber MWS warning, RUDDER BIAS is illuminated when either RUDDER BIAS switch (A or B) is set to the OFF position.

Should a leak occur in one supply line to a strut, the resulting loss of air would create an imbalance of forces which would bias the rudder to one side.



RUDDER BIAS

Figure 15-6. Rudder Bias Switches

Selecting the RUDDER BIAS switch on the affected side to OFF equalizes the pressures on both sides of the strut.

The strut on the good side then maintains a balance of forces on the rudder.

Electric Heating

An electric heater muff on each strut ensures that ice will not prevent operation of the strut or solenoid valve. Each muff has two elements which are supplied separately from the L and R PITOT/VANE HEAT switches on the roof panel (Figure 15-7).

Each switch controls the power supply to one heating element of each strut. See also Table 15-1.

GUST LOCK SYSTEM

General

An aileron/elevator gust lock, attached to the airplane structure, when connected to the left control wheel, prevents aileron and elevator movement. Throttle lever movement is also restricted.

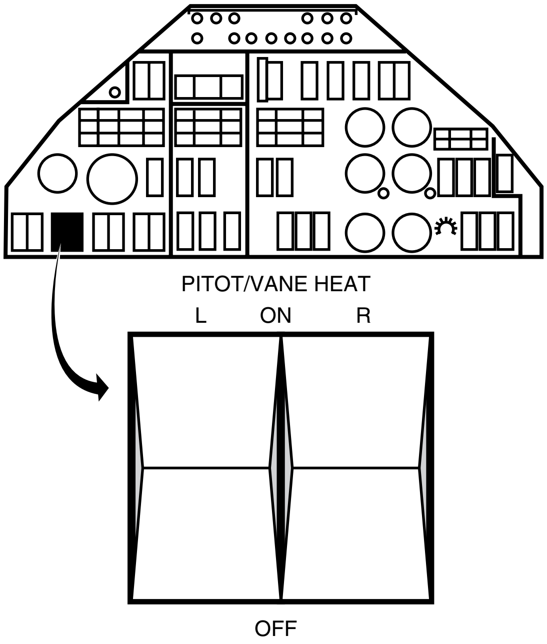


Figure 15-7. PITOT/VANE HEAT Switches

Table 15-1. ELECTRICAL SOURCES—PITOT/VANE HEAT

PITOT VANE SWITCH	BUSBAR	CIRCUIT BREAKER
LEFT	PS1	RUD BIAS HEAT LT DA-D E/8
RIGHT	PS2	RUD BIAS HEAT RH DA-D F/8

A tubular rudder gust lock, when inserted through a hole in the cockpit floor, engages rudder linkage to immobilize the rudder.

Operation

To engage the aileron/elevator gust locks, disengage the lock from the stowage clip and swing it in position with the neutralized control wheel (Figure 15-8). Position the claws on the control wheel bushings. Push the knurled knob down and rotate clockwise to engage the claws in the bushing.

A cable actuated by aileron/elevator gust lock engagement actuates the throttle lockout (baulk unit) mechanism. This restricts throttle movement, allowing either throttle to be moved full range while the other throttle is restricted to a low power setting. When the gust lock is removed, the baulk is removed, freeing both throttle levers. The lock must be removed prior to takeoff.

To engage the rudder gust lock (Figure 15-8), lift the cover in the floor and remove the rudder gust lock from stowage at the front side of the copilot's seat. Center the rudder and insert the lock through the hole in the rudder linkage and into the bracket beneath. The lock must be removed prior to taxiing.

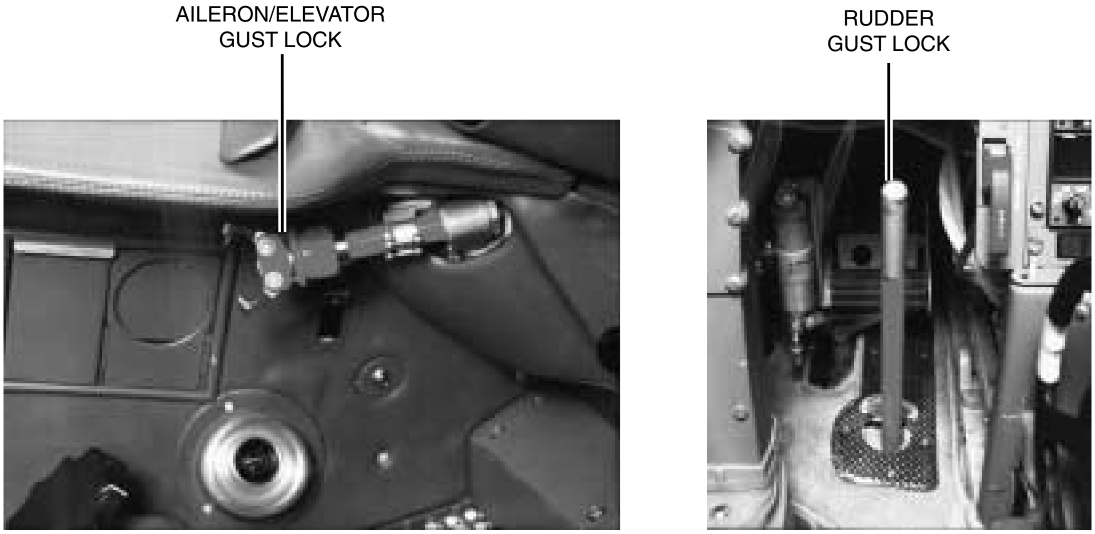


Figure 15-8. Gust Lock System

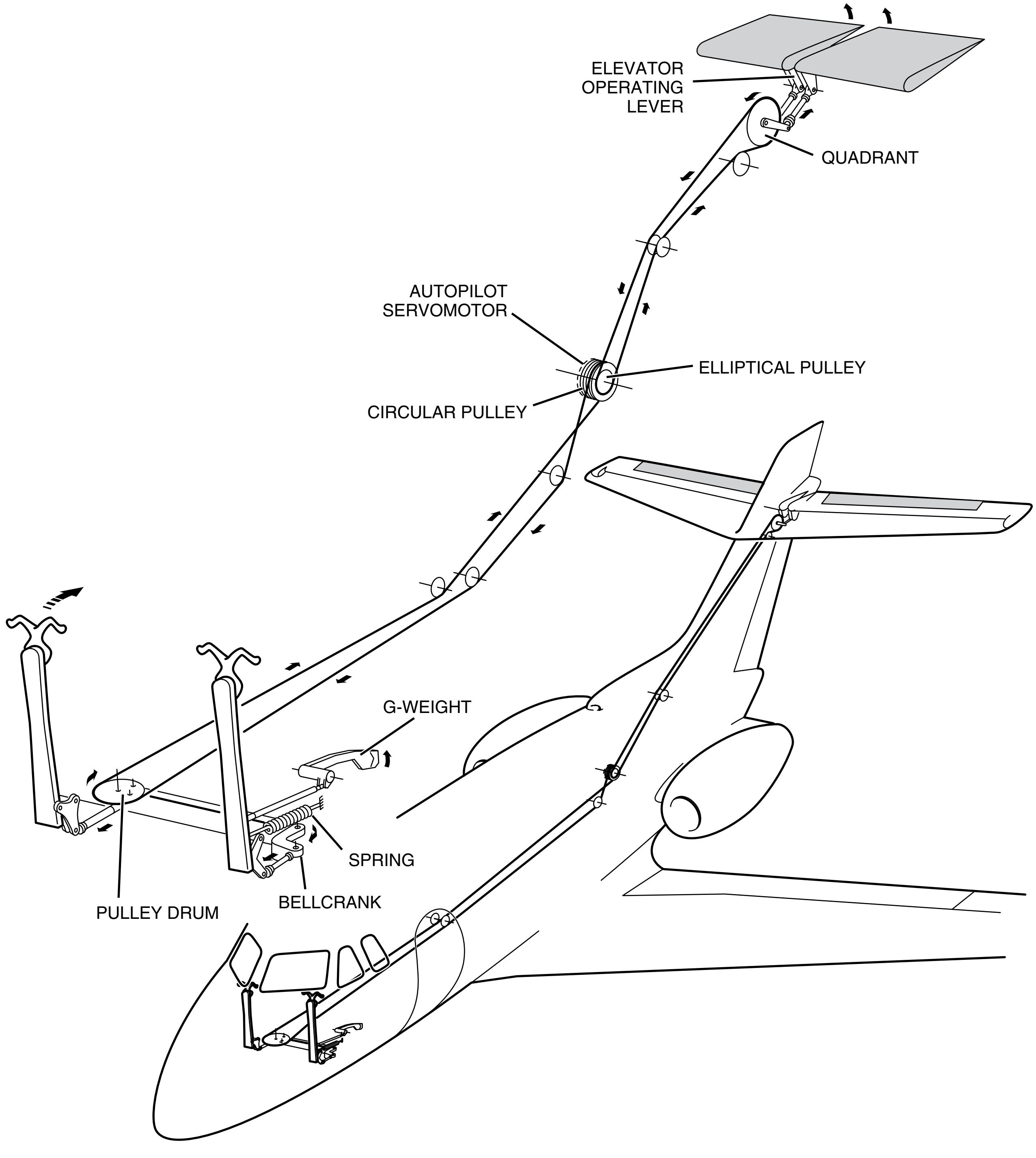


Figure 15-9. Elevator System

ELEVATOR CONTROL SYSTEM

The elevator control surfaces are hinged from the rear of the horizontal stabilizer. Each is fitted with a mass balance weight to prevent flutter and a horn balance to reduce the aerodynamic loads felt by the pilot (Figure 15-9).

A spring and a G-weight are connected to the left control circuit to provide acceptable stick force characteristics.

A stick shaker motor is attached to each control column and is operated by the stall warning system. A stick pusher unit is connected to the elevator cable system to provide a positive push forward (pitch down) at the onset of stall identification. See section titled, *Stall Warning and Identification System*.

The cable circuit also provides an input for the autopilot servo unit.

Primary stops controlling the range of elevator movement are located on the middle hinge of each control surface. Secondary stops are located at the pulley drums under the flight compartment floor.

Elevator Trim Control System

A trim tab is fitted to each elevator and is manually controlled from an elevator trim handwheel on the central control pedestal (Figure 15-10). Operation of the elevator moves the trim tab to give servo action.

The autopilot servomotor is used to provide pitch trim, Mach trim and electric trim functions.

If necessary these functions can be overridden by arresting the movement of the elevator trim handwheel. The handwheel turns as the automatic system applies trim via the servo.

Elevator Out-Of-Trim Warning

When an elevator is out of trim at takeoff, a warning is given by the ELEV/AIL TRIM annunciator illuminating on the main MWS panel (Figure 15-10) if the following conditions apply:

- Aircraft weight is on wheels.
- Both engine thrust levers are advanced to more than 40–60% N_1 power.

Mach Trim

The Mach trim system uses a Mach trim computer and the same pitch trim servomotor as the manual electric trim and the autoelectric trim systems. The Mach trim computer accepts Mach number data from the air data computers and computes trim tab deflection, increasing longitudinal stability at Mach numbers above 0.73 Mach when the autopilot is not engaged.

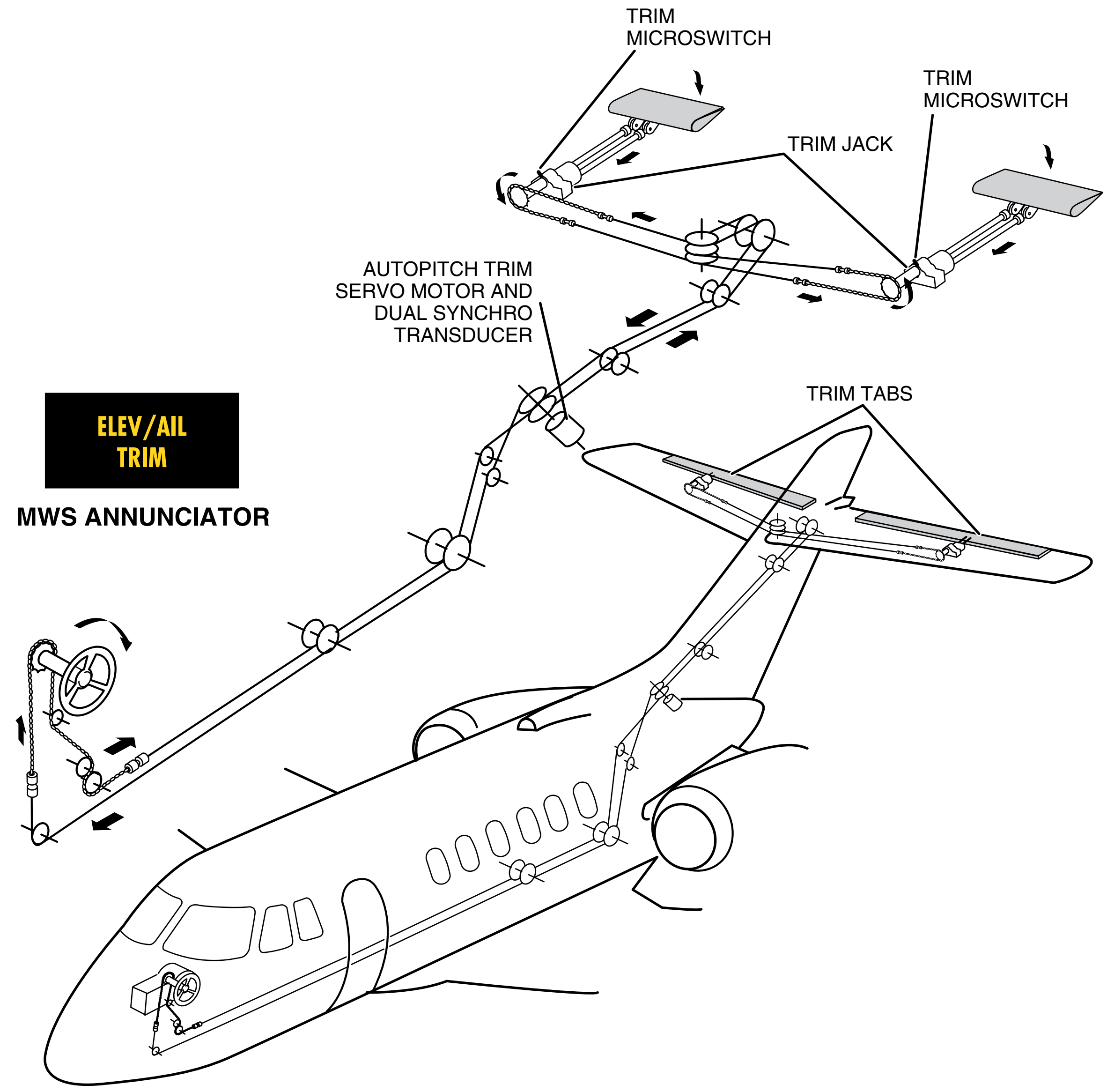


Figure 15-10. Elevator Trim System

The Mach trim computer is mounted underneath the copilot's seat.

There are no power switches for the system. On power-up of the aircraft busbars, the Mach trim system performs a self-test routine. Providing the following conditions are met, the system engages:

- Indicated Mach number is less than 0.73M.
- The autopilot is not engaged.
- The electric trim is not in use.
- The manual pitch trim handwheel is not being used.
- No Mach trim system failures have been detected.
- Power supplies are correct and available to the system.

Faults which render the system inoperative will cause the system to disconnect in a fail passive manner and will result in a warning given by the MACH TRIM FAIL annunciator illuminating at the top right of the MWS panel (Figure 15-11).

The MACH TRIM FAIL annunciator and the Mach trim computer output to the annunciator can be tested during preflight by pushing the ADC AHRS 1 & 2 self-test switches (Figure 15-11).

Satisfactory operation will be indicated by the MACH TRIM FAIL annunciator coming on while a switch is operated.

Electric Power Supplies

The ELEV/AIL TRIM annunciator is powered from the PE busbar.

The Mach trim system receives 28-VDC power from the PS1 and PS2 busbars through two circuit breakers located on subpanels DA-D (Figure 15-12):

- MACH TRIM PWR, DA-D C/9, busbar PS1
- MACH TRIM MON, DA-D D/2, busbar PS2

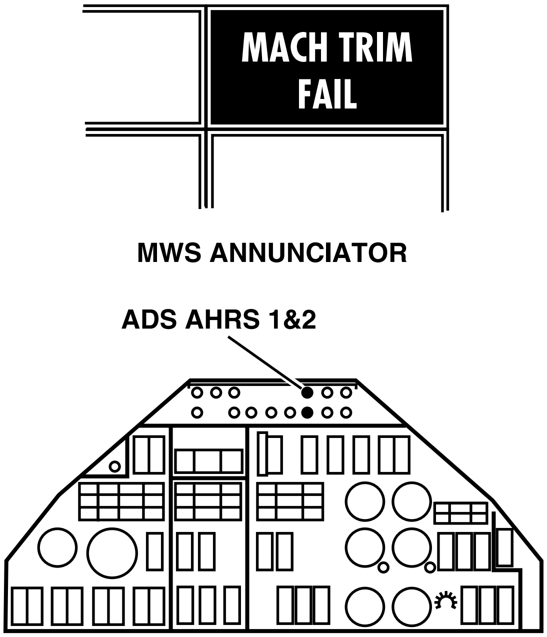


Figure 15-11. MACH TRIM FAIL Annunciator and ADS AHRS 1 & 2 Self-Test Switches

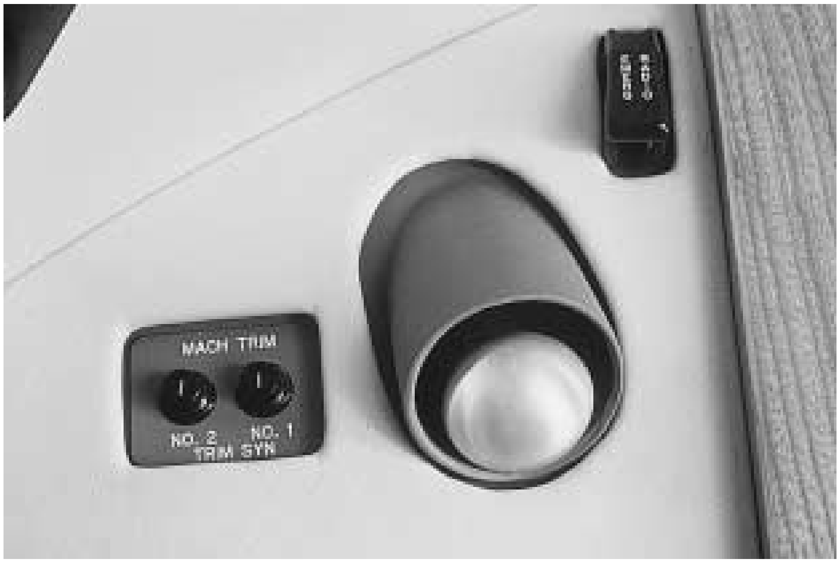


Figure 15-12. Mach Trim Circuit Breakers

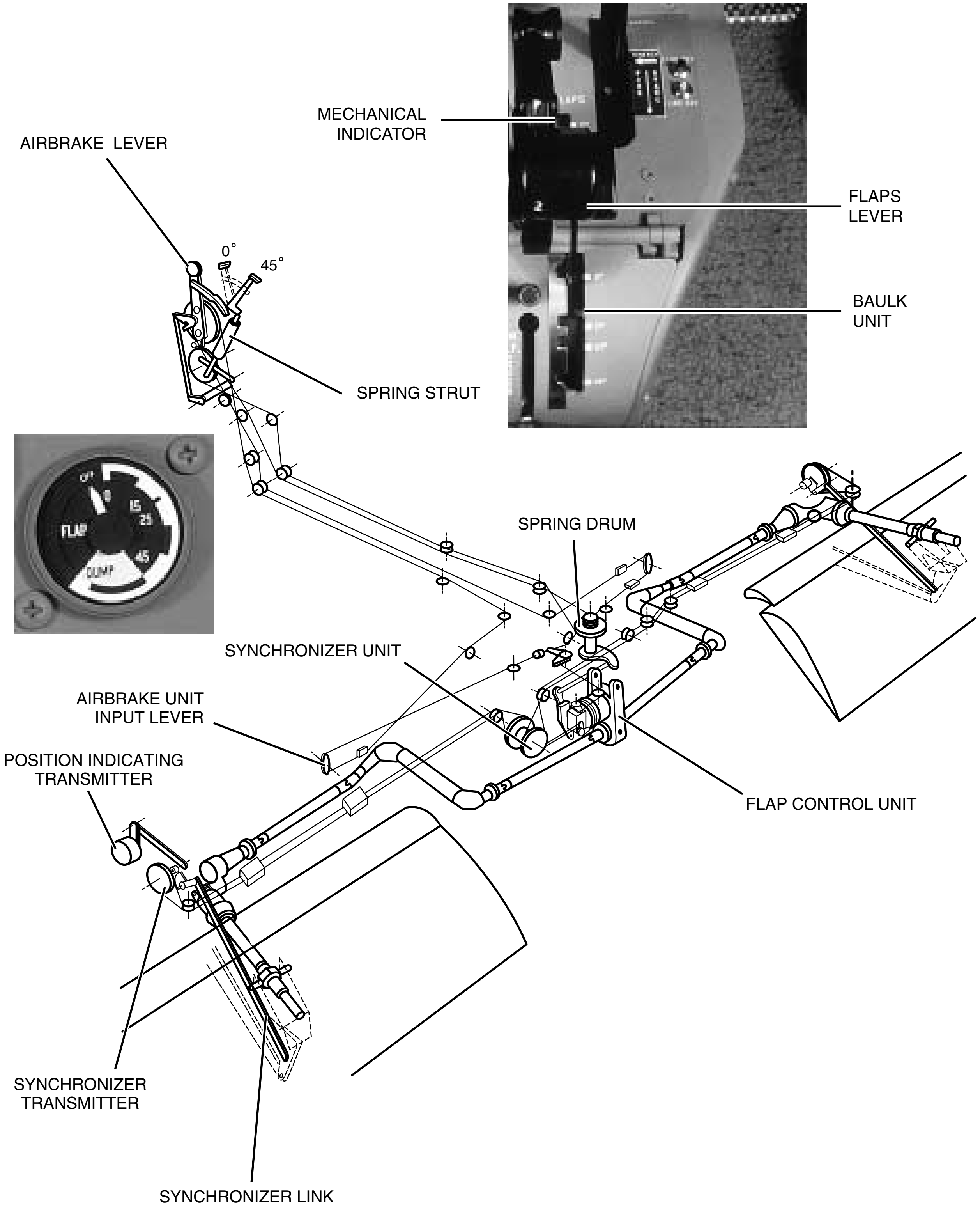


Figure 15-13. Flap Control System Diagram

SECONDARY CONTROLS

FLAP SYSTEM

General

Interconnected slotted flaps are hydraulically powered from a single flap control unit and transmission shafting (Figure 15-13).The flap control unit is normally supplied from the main hydraulic system, but an independent fluid supply from the auxiliary hydraulic system is available in the event of a main system failure.

The flaps are controlled by a selector lever located on the right side of the central control pedestal (Figure 15-14). The lever moves in a gated slot which corresponds with the following flap positions:

- 0° (up or takeoff)
- 15° (takeoff)
- 25° (approach)
- 45° (landing)

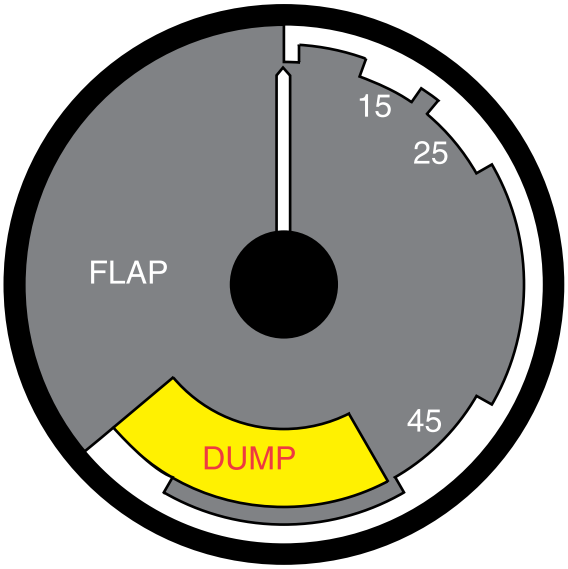
A baulk, operative at the 15° slot, prevents inadvertent selection of the up, approach or landing positions. The baulk is released by pushing the selector lever into the 15° slot in the gate.

Additional movement of the flaps downwards, from the landing position to a lift-dump position, is controlled by the airbrake selector lever.

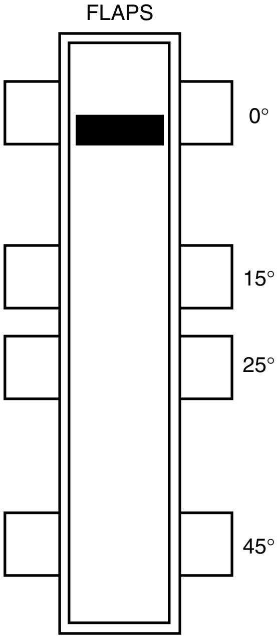
Indications and Warnings

Flap position is indicated on an electrical indicator located on the right side of the center instrument panel (Figure 15-14). This indicator also shows the lift-dump position.

A mechanical indicator located adjacent to the thrust levers shows the position of the flap selector lever. Additionally, the flap positions are marked against the corresponding gates of the slot in which the lever moves.



FLAP POSITION INDICATOR



FLAP SELECTOR LEVER

Figure 15-14. Flap Selector Lever Position Indication

A warning horn sounds if the flaps are selected to the 25 or 45° positions when the landing gear is not locked down.

Flap Control System

The flap selector lever is connected by a cable to an input lever on the flap control unit. A spring strut and a spring drum in the cable run compensate for any movement lag (Figure 15-13).

Movement of the input lever connects hydraulic pressure to operate a hydraulic motor in the flap control unit. The motor drives the transmission shafting to move the flaps in the selected direction. The motor is stopped automatically should a control cable failure occur.

When the flaps reach the selected position, the hydraulic pressure is removed from the motor. Further movement of the flaps is arrested and they are locked at the required angle. Friction devices prevent the flaps moving away from the selected position in the event of hydraulic failure.

A synchronizing cable circuit is provided to prevent the occurrence of an asymmetric flap condition. Should one flap cease to move, the synchronizing circuit isolates the hydraulic supply to the flap motor, the flaps stop moving and maintain the position attained at the moment of failure.

A flap servo baulk stops additional movement of the flap control unit input lever from the landing position. The baulk is released and the lever moved to the lift-dump position by an interconnection with the airbrake control when this is moved into the lift-dump position.

Airbrakes

In flight, the airbrakes must not be operated when the flaps are extended to any position.

Lift Dump

Lift dump is to be used only when the aircraft is on the ground.

Rudder Bias

The rudder bias switches must be ON and the systems operative during each takeoff.

Electrical Power Supplies

The flap position indicator is powered from the PS1(a) busbar via circuit breaker FLAP POS (M/7) on panel DA-D.

The warning horn is powered from the PE(a) busbar via circuit breaker HORN (K/1) on panel DA-D.

Emergency Operation

The flaps may be operated by means of the auxiliary hydraulic system. Using the auxiliary system, the flaps can be lowered to any preset position from 0 to 45°. Lift dump and airbrakes are not available.

The emergency flap selection is made using the normal flap control lever. Position indication is still available as long as PS1(a) busbar is energized.

AIRBRAKE CONTROL SYSTEM

A pair of airbrakes per wing are powered by the main hydraulic system (Figure 15-15). Of each pair, one airbrake extends from the wing upper surface, the other one from the wing lower surface.

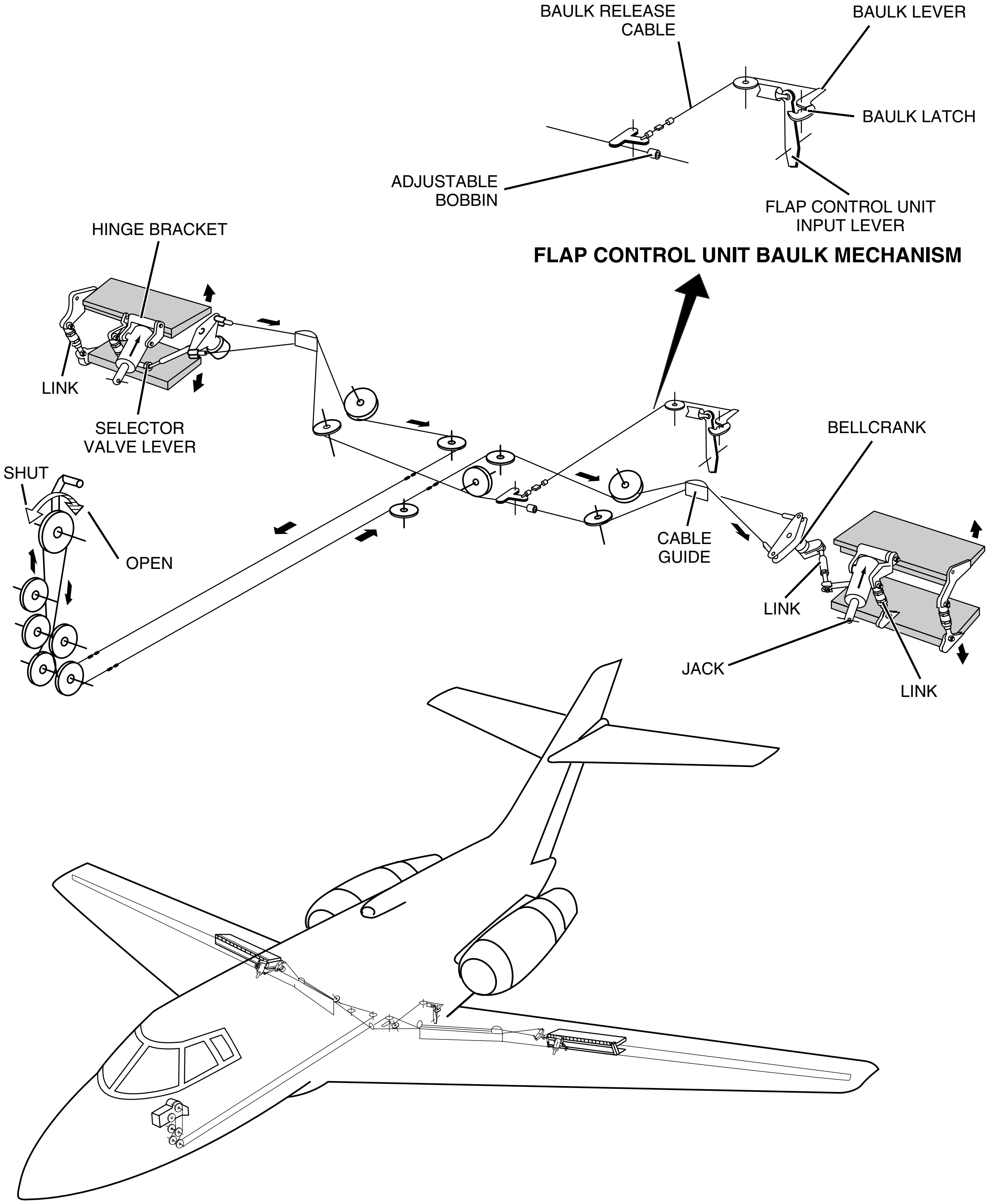


Figure 15-15. Airbrake Control System

The airbrakes are controlled by an AIRBRAKE selector lever located on the center control pedestal (Figure 15-16). The lever moves ina slot having a scale marked with three positions: SHUT, OPEN, and DUMP.

The normal (flight and ground) operating range is from the SHUT and fully OPEN selections. Intermediate positions are obtained by leaving the selector lever at the required setting within the range.

A system of cables run from AIRBRAKE selector lever to a hydraulic selector valve mounted on each airbrake actuator. Main hydraulic system pressure is used to operate the airbrakes through a hydraulic actuator in each wing. To cater for control system failure, the selector valve is biased to the SHUT position.

There is no service available from the auxiliary hydraulic system for the airbrakes should the main hydraulic system fail.

Lift Dump

The AIRBRAKE selector is interconnected with the input lever to the flap control unit. Selection of airbrake between the SHUT and OPEN positions does not affect the flap position.

During the landing run with the flaps selected to the 45° (land) position, lifting the AIRBRAKE selector and then moving it rearwards into the DUMP position automatically lowers the flaps from the 45° setting to 75°. It also opens the airbrakes further to provide maximum drag.

Annunciations and Warnings

The position of the airbrakes is shown by an AIR BRAKES OPEN-SHUT magnetic indicator. The indicator is connected to two microswitches, one on each airbrake. With electrical power off, cross-hatching is displayed. With power on, the indicator shows OPEN when either airbrake is in any position other than fully closed. With both airbrakes closed, SHUT is displayed.

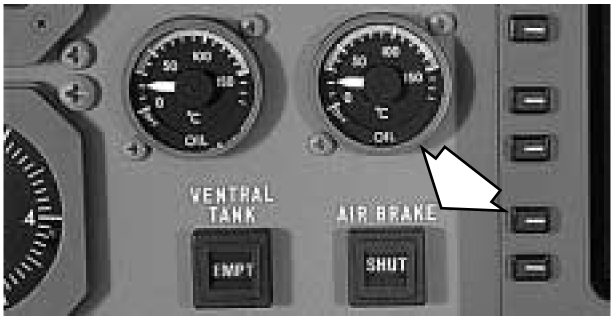
A white AIR BRAKES annunciator is illuminated on the MWS when the airbrakes are extended from the SHUT position (Figure 15-16).

A warning horn sounds if all of the following conditions occur:

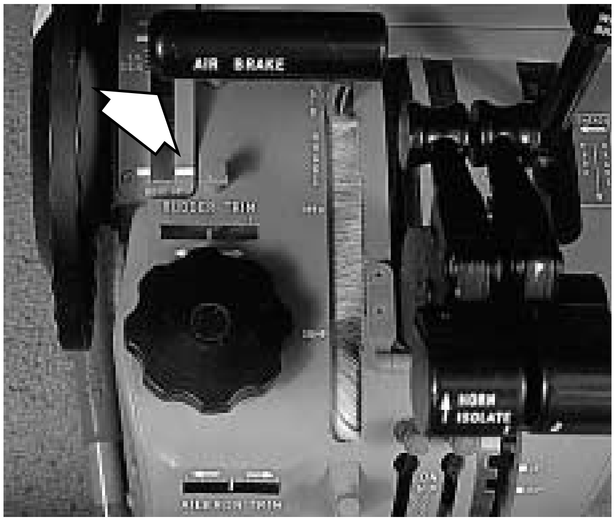
- The landing gear is down.
- Both thrust levers are advanced to more than 40–60% N_1 power.
- The airbrakes are extended from the SHUT position.

Electrical Power Supplies

The AIR BRAKES annunciator is powered from PS2(a) busbar via circuit breaker AIRBRAKE (L2) on panel DA-D.



OPEN/SHUT ANNUNCIATOR



AIR BRAKE SELECTOR LEVER

Figure 15-16. Airbrake System Control and Indication

STALL WARNING AND IDENTIFICATION SYSTEM

GENERAL

A stall warning and identification system is provided to emphasize the aircraft's natural cues available at the point of stall. The system functions are:

- Stick shaker (warning)
- Stick pusher (identification)

The system comprises two sensing channels, each using an airflow angle sensor vane, a signal summing unit (SSU), and a third sensing channel which uses two underwing pressure-sensing vents and a stall identification sensor.

Warning and identification is provided by two stick shaker motors, an hydraulically operated actuator with two integral electrohydraulic valves respectively, and associated logic, annunciators and test switches.

System Logic

1. It is impossible for a stick push to occur before a stall warning (stick shake).
2. No single active fault of an SSU or relay can cause the operation of a stall valve or the associated red STALL VALVE annunciator.
3. The autopilot is disengaged when a stall warning signal is initiated. This prevents the autopilot from attempting to counteract the resulting stick shake operation or subsequent stick push.

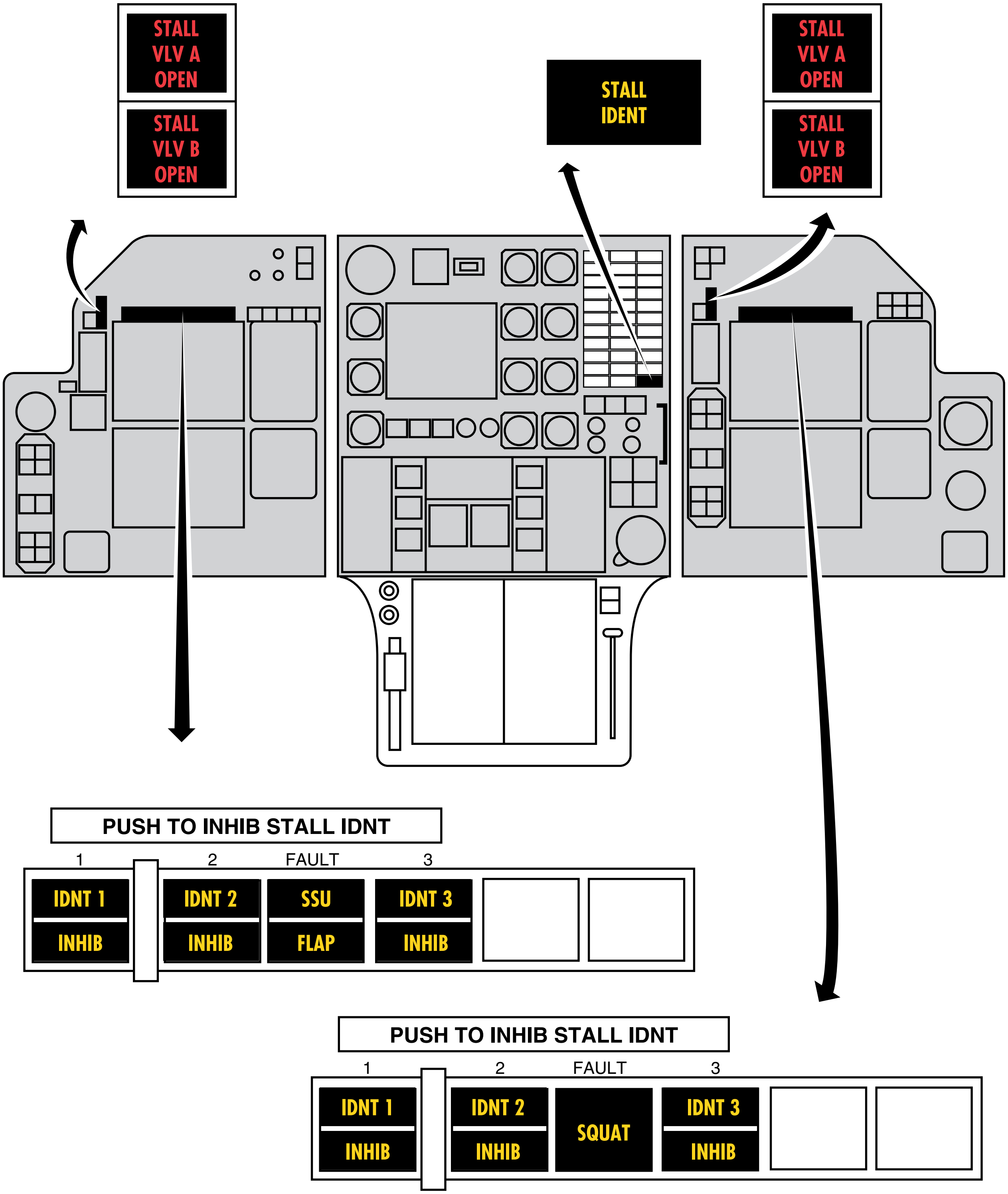


Figure 15-17. Stall Warning and Identification System—Controls and Annunciations

Controls and Annunciations

Figure 15-17 shows the controls and annunciations on the main instrument panels associated with the stall warning and identification system.

System faults are indicated on two groups of amber annunciators, one per pilot.

Three STALL switches are located in the TEST section of the roof panel (Figure 15-18).

Anti-icing heating of the airflow sensor vanes is controlled from the PITOT/VANE HEAT switches (Figure 15-18).

Indication of vane heater failure is provided by the two amber L and VANE HTR FAIL annunciators also on the roof panel and by the ICE PROT ↑ repeat annunciator on the MWS panel (Figure 15-18).

There are no control switches. The stall warning part of the system becomes armed on takeoff (no weight-on-wheels), while the stall identification part of the system becomes armed 6 seconds after takeoff.

SYSTEM DESCRIPTION

Refer to Figure 15-19 for a schematic of the system in normal conditions.

The system has two channels for stall warning and identification and a third channel which provides stall identification output only. Normal function of the complete stall warning and identification system is shown in the flow charts presented in Figure 15-20.

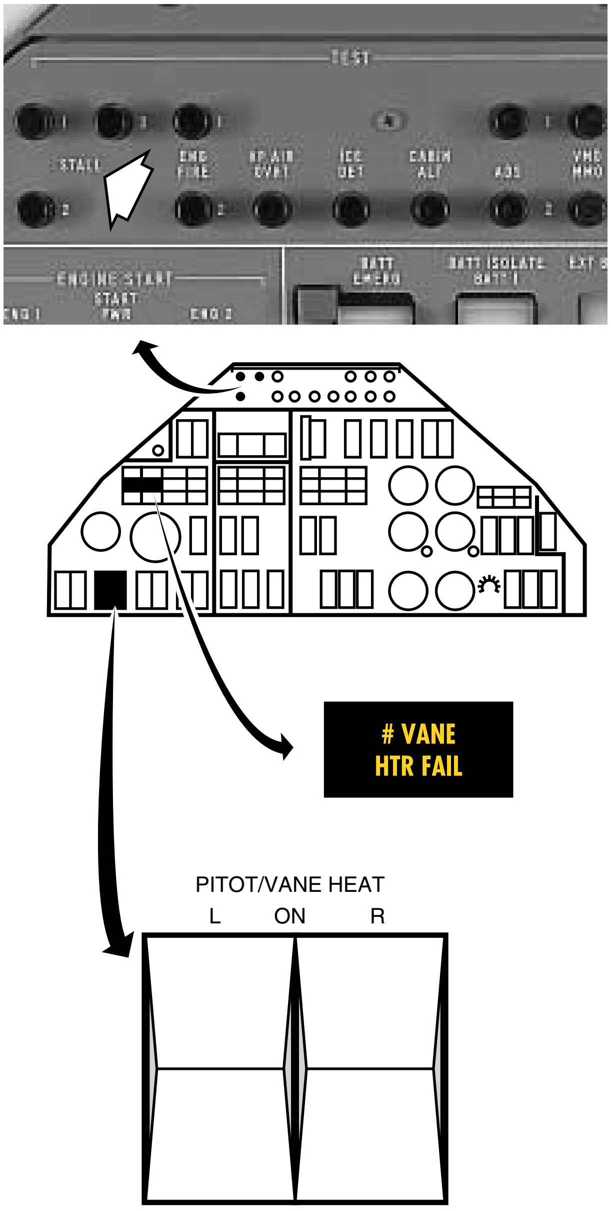


Figure 15-18. System Test and Anti-icing Control and Indication

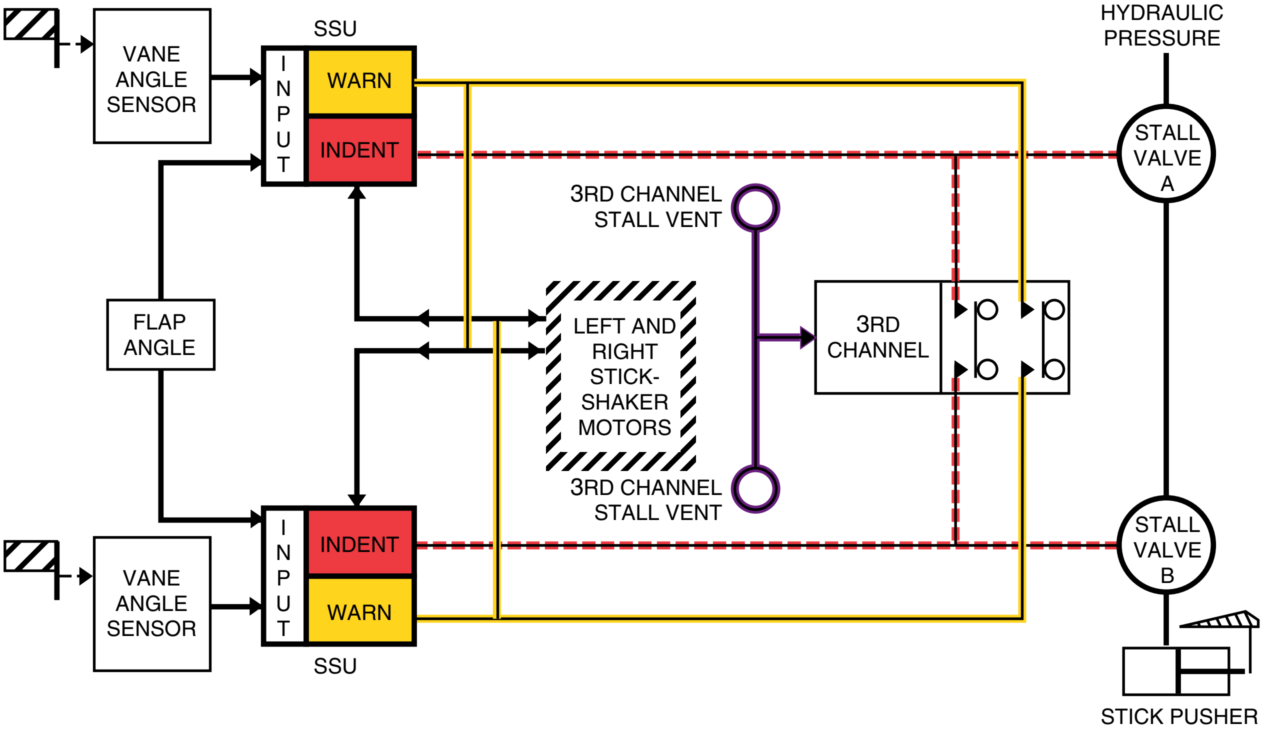


Figure 15-19. Stall Warning and Identification System Schematic

Stall Warning System

The stall warning system uses an electrically driven stick shaker on each control column to provide a physical warning of an approaching stall to the pilots.

Angle-of-attack is derived from two airflow angle sensor vanes (Figure 15-21) mounted one on each side of the forward fuselage. Electrical signals proportional to vane angle are sensed in the associated SSU which also receives inputs relating to flap angle. From the flap angle signal, the SSU calculates the point of stall warning. When the vane angle corresponds to that point, the SSU provides an output to operate the stick shaker motor on each control column.

The stall warning system is inhibited while the aircraft weight is on the wheels to prevent wind gusts triggering false stick shaker operations on the ground. At takeoff, the system is armed and commences to monitor the pitch attitude of the aircraft.

Stall Identification System

The stall identification system uses a hydraulic stick pusher to force the control column forward (pitch down) at the calculated point of stall. The stick pusher is powered by main hydraulic system pressure (backed by the main accumulator). The rate of operation is controlled by a fluid restrictor. Control of the hydraulic pressure to the stick pusher is via two independent stall valves (A and B), connected in series and mounted integrally with the stick pusher. Both stall valves must be open to activate the unit.

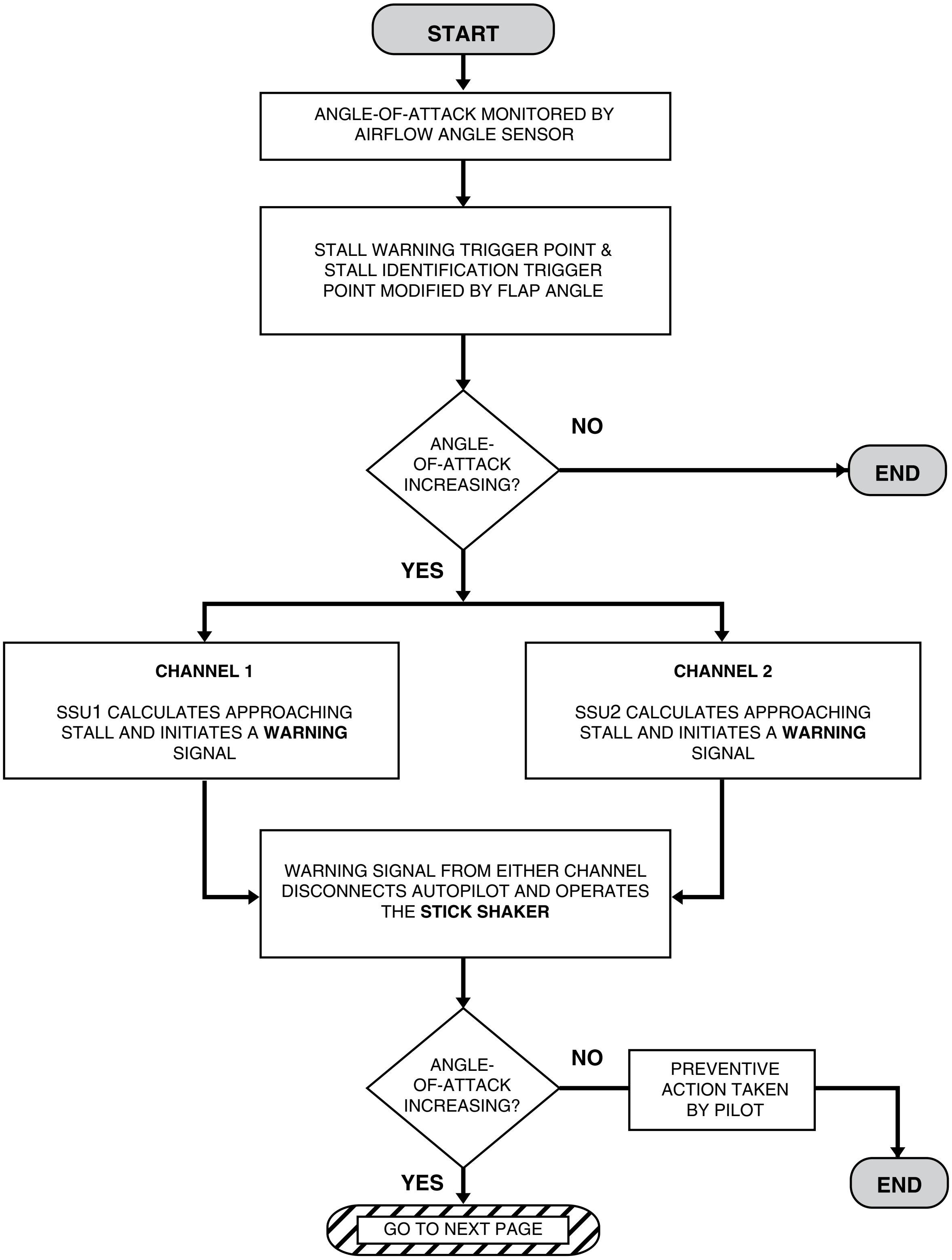

START
ANGLE-OF-ATTACK MONITORED BY AIRFLOW ANGLE SENSOR
STALL WARNING TRIGGER POINT & STALL IDENTIFICATION TRIGGER POINT MODIFIED BY FLAP ANGLE
ANGLE-OF-ATTACK INCREASING?
NO
END
YES
CHANNEL 1
SSU1 CALCULATES APPROACHING STALL AND INITIATES A WARNING SIGNAL
CHANNEL 2
SSU2 CALCULATES APPROACHING STALL AND INITIATES A WARNING SIGNAL
WARNING SIGNAL FROM EITHER CHANNEL DISCONNECTS AUTOPILOT AND OPERATES THE STICK SHAKER
ANGLE-OF-ATTACK INCREASING?
NO
PREVENTIVE ACTION TAKEN BY PILOT
END
YES
GO TO NEXT PAGE

Figure 15-20. Stall System Flow Logic (Sheet 1 of 2)

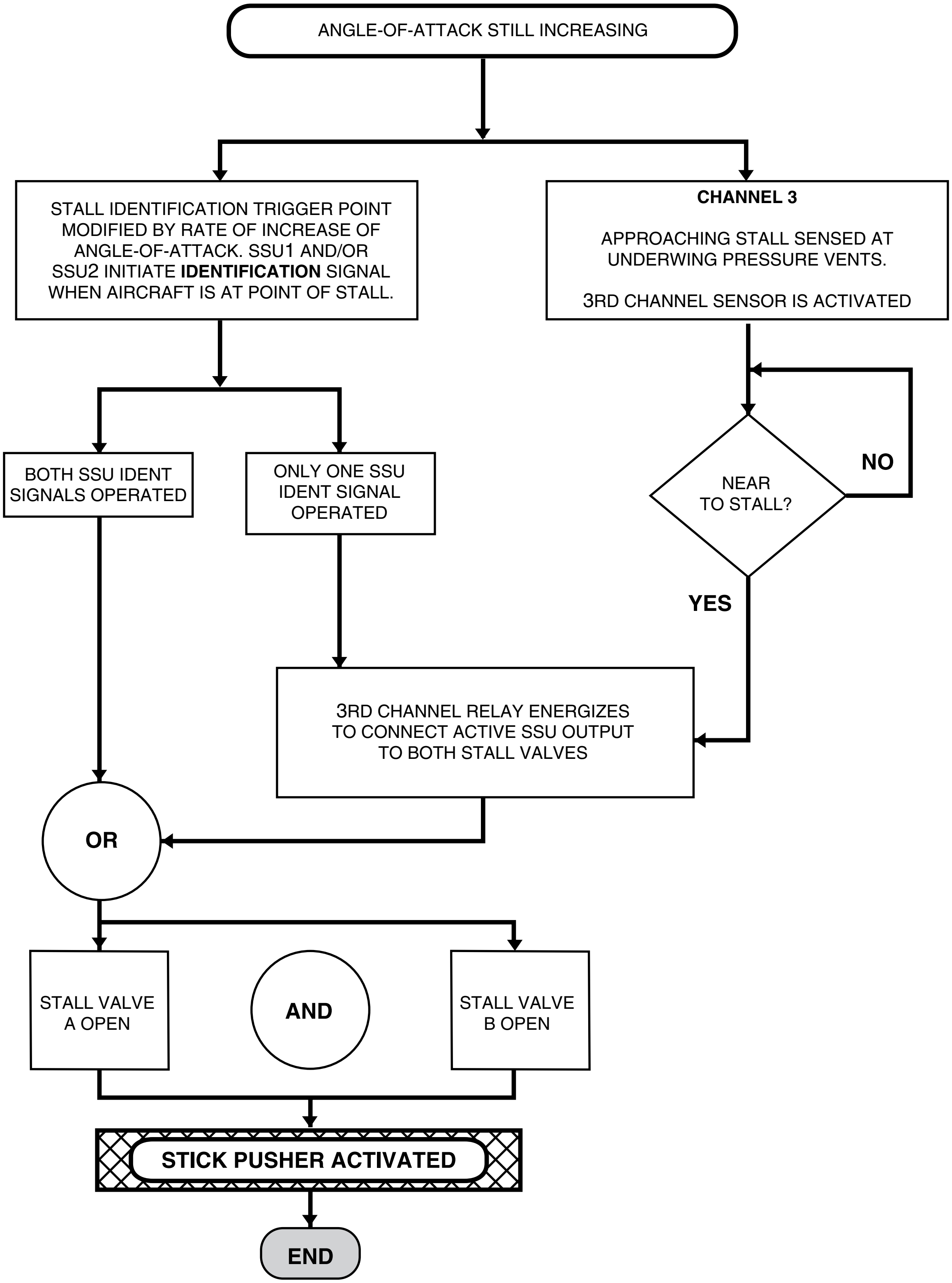

ANGLE-OF-ATTACK STILL INCREASING
STALL IDENTIFICATION TRIGGER POINT MODIFIED BY RATE OF INCREASE OF ANGLE-OF-ATTACK. SSU1 AND/OR SSU2 INITIATE IDENTIFICATION SIGNAL WHEN AIRCRAFT IS AT POINT OF STALL.
CHANNEL 3
APPROACHING STALL SENSED AT UNDERWING PRESSURE VENTS.
3RD CHANNEL SENSOR IS ACTIVATED
BOTH SSU IDENT SIGNALS OPERATED
ONLY ONE SSU IDENT SIGNAL OPERATED
NEAR TO STALL?
NO
YES
3RD CHANNEL RELAY ENERGIZES TO CONNECT ACTIVE SSU OUTPUT TO BOTH STALL VALVES
OR
STALL VALVE A OPEN
AND
STALL VALVE B OPEN
STICK PUSHER ACTIVATED
END

Figure 15-20. Stall System Flow Logic (Sheet 2 of 2)

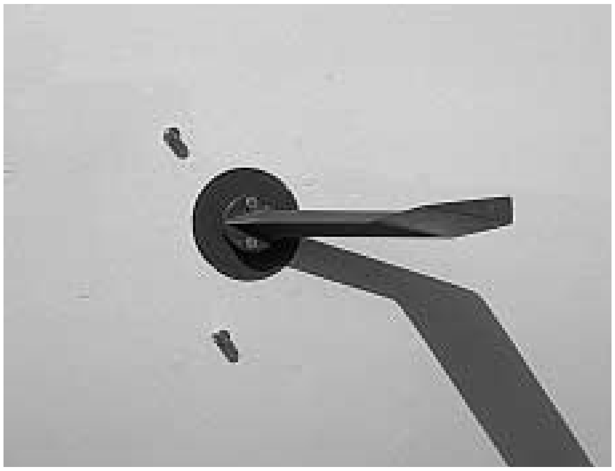


Figure 15-21. Airflow Angle Sensor

The output from one SSU energizes one stall valve. An associated red STALL VLV A or B OPEN annunciator is illuminated to indicate that the valve is being signaled to open (Figure 15-22).

The SSU uses the vane sensor and flap angle inputs to calculate the point of stall. However, another factor, the rate of vane angle increase, may also modify the calculation (Figure 15-23). For example, the point of stall is advanced when the rate of increase is high (dynamic stall). When the vane angle agrees with the calculated point of stall, the SSU produces an output (stall identification) to energize the associated stall valve and annunciator.

The system is inhibited when the aircraft is on the ground by the weight-on-wheels switch. To prevent an unwanted stick push from occurring until flight is assured, the stall identification part of the system remains inhibited from takeoff until a 6-second timer within the SSU times out.

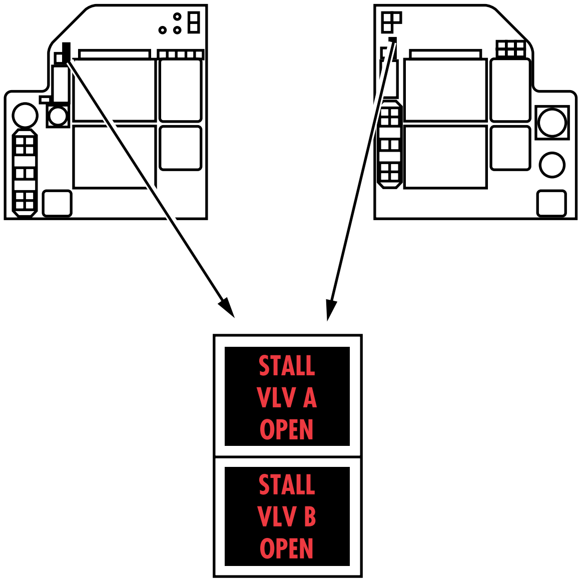


Figure 15-22. STALL VALVE A/B Annunciators

When an SSU produces a stall warning output, which is then followed by an identification signal, the warning signal latches the stall identification relay for the same channel. The latch is removed when the warning output ceases. This ensures that the stick push is maintained until the aircraft has reached nose-down attitude well below the stall point, and stops the system hunting around the limited hysteresis at the stall identification setting.

Third Stall Identification Channel

A third channel for stall identification is provided by a pitot/static system which uses pitot pressure (P_2) from the right pitot head, static pressure (S_6) from the forward static vents and a vent pressure (V) from left and right stall vents. The stall vents are located on the underside of the left and right wings.

These pressures are sensed by a capsule-operated stall detector. This is set to produce an output at a point between the settings for the stall warning and identification signals from the SSUs.

The output from the third channel sensor energizes a relay which connects the stall identification output from one channel's SSU to the stall valve of the other channel. Thus, with the third channel output activated, both stall valves A and B will open following a stall identification output from only one SSU. This ensures system integrity should a SSU fail.

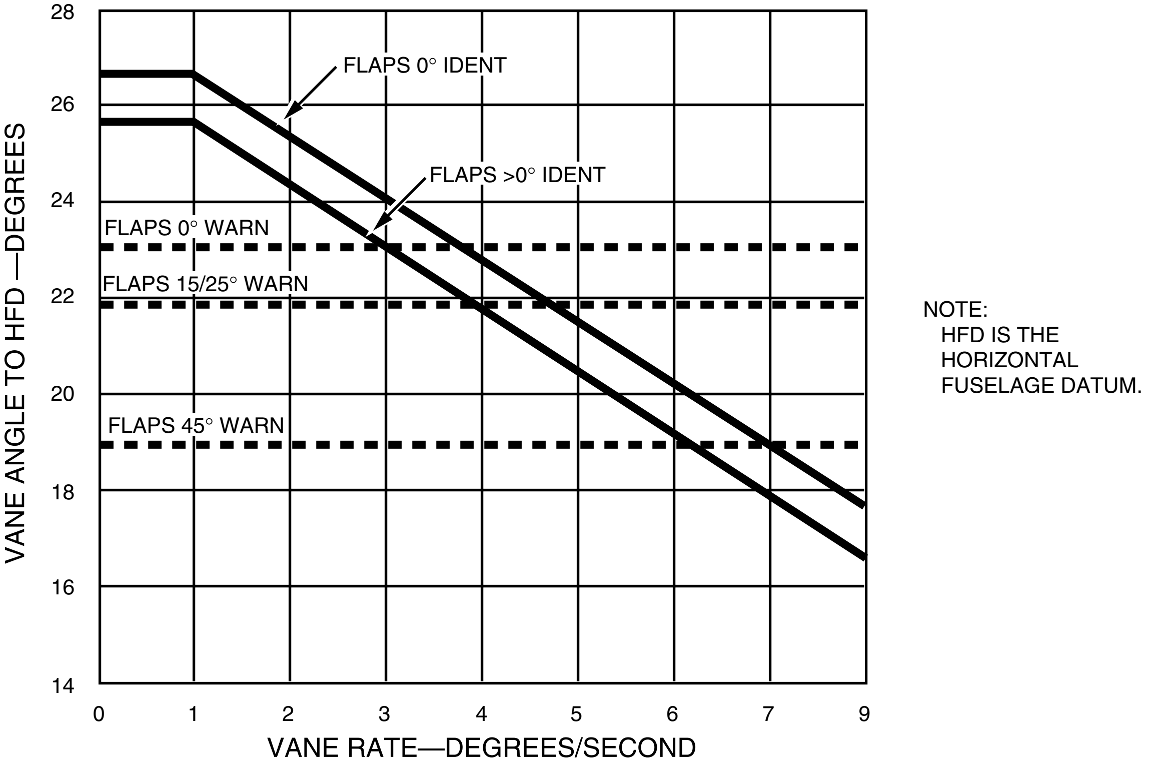


Figure 15-23. Variation of Stall Identification Angle With Increased Vane Angle

System Faults and Annunciations

The duplication of the stall warning system ensures that a single passive fault cannot prevent a stick shake from occurring. Should an active fault develop, the faulty system can be isolated via the appropriate circuit breaker. The remaining good system will still operate both stick shaker motors.

The stall identification system is designed so that a single active fault cannot give an inadvertent stick push. At the same time, the system also ensures that a single passive fault would not prevent a push operation from occurring, when required.

All annunciator warnings in the stall warning and identification system will also cause a repeater STALL IDENT annunciator on the MWS panel to illuminate (Figure 15-24).

The power to energize a stall valve is routed via the identification relay of one channel and the warning relay of the other channel.

A monitoring circuit will cause an IDNT 1 or IDNT 2 annunciator (depending on the channel at fault) to illuminate after a 4-second time delay—if an identification signal from one SSU has been triggered without a warning signal from the other SSU. The IDNT 1 or IDNT 2 annunciators are part of the IDNT/INHIB switches provided for both pilots (Figure 15-24). When an IDNT annunciator is illuminated, either pilot can push the associated switch to inhibit the faulty channel. The INHIB annunciator part of the switch will then illuminate.

The third channel sensor, together with the remaining SSU, would provide a stick push operation when required.

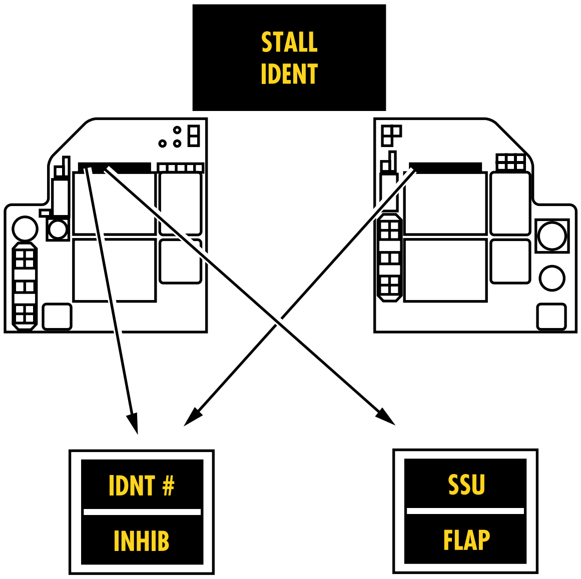


Figure 15-24. Stall System Annunciators

The pilots may attempt to reset the failed channel by operating an INHIBIT RESET switch (Figure 15-25). This switch is located on the stall diagnostic panel (inboard edge of the pilot's bulkhead). Should the channel fail again after reset, it should be inhibited and left in that condition for the remainder of the flight.

Flap Asymmetry

If asymmetry between the left and right flap positions exists for more than 2 seconds, then the FLAP annunciator (pilot's group only) will illuminate.

Weight-On-Wheels Asymmetry

If asymmetry between the positions of the left and right weight-on-wheels relays exists for more than 4 seconds, then a SQUAT annunciator (copilot's group only) will illuminate.

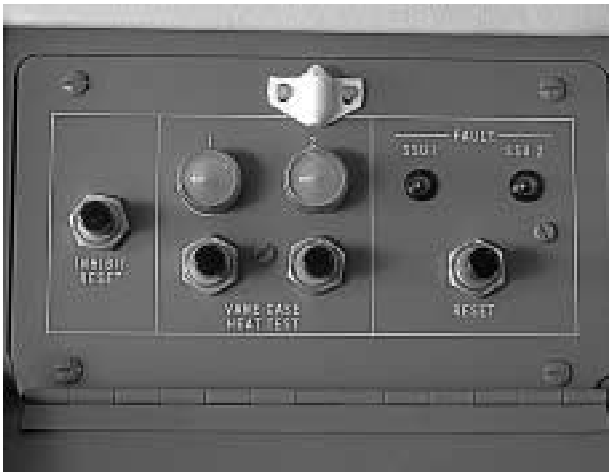


Figure 15-25. Stall Identification and Diagnostic Panel

SSU Self-Test

A built-in test within the SSU detects the following faults:

- Loss of 26-VAC supply
- Loss of internal power or short circuit
- Loss of airflow angle sensor excitation
- Loss of flap position input
- Airflow angle sensor transformer winding open or short circuit

If any of the above occur, an external relay causes a SSU annunciator to illuminate. If the weight-on-wheels switch is in the flight condition, a magnetic indicator associated with the faulty SSU will display "white." The magnetic indicators are located on the stall identification diagnostic panel (Figure 15-25).

Electrical Supplies

The stall warning and identification system uses both DC and AC power (Table 15-2).

Where circuit breakers are used to protect the circuits, these are located on panel DA.

Table 15-3 provides a summary of the annunciators and their functions.

Table 15-2. ELECTRICAL SOURCES—STALL/INDENTIFICATION SYSTEM

PANEL LOCATION ROW/COLUMN	CIRCUIT BREAKER	CIRCUIT OR EQUIPMENT	BUSBAR
DA-D B/1	STALL IDENT 1	26-VAC INPUT TO SSU 1	XS 1
DA-D B/4	STALL VLV A	28-VDC TO STALL VALVE 1 AND ANNUNCIATORS	PS1(a)
DA-D B/6	STALL WARN MOTOR 1 (LH)	28-VDC TO STICK SHAKER MOTOR 1	PS1(a)
DA-D B/2	STALL IDENT 2	26-VAC INPUT TO SSU 2	XS2
DA-D B/5	STALL VLV B	28-VDC TO STALL VALVE 2 AND ANNUNCIATORS	PS2(a)
DA-D B/7	STALL WARN MOTOR 2 (RH)	28-VDC TO STICK SHAKER MOTOR 2	PS2(a)
DA-D B/3	STALL IDENT 3	28-VDC TO SSU 3	PE

Table 15-3. STALL SYSTEM ANNUNCIATOR FUNCTIONS

FUNCTION	ANNUNCIATOR		MWS PANEL
AIRFLOW ANGLE SENSOR LEFT OR RIGHT VANE HEATING MALFUNCTION	L (R) VANE HTR FAIL	# VANE HTR FAIL	ICE PROT ↑
FLAP POSITION ASYMMETRY	FLAP	FLAP	STALL IDENT
STALL IDENTIFICATION CHANNEL 1 (2) (3) FAULT	IDNT 1 (2) (3)	IDNT #	
SIGNAL SUMMING UNIT FAULT	SSU	SSU	
WEIGHT-ON-WHEELS ASYMMETRY	SQUAT	SQUAT	
STALL VALVE A (B) OPERATING	STALL VLV A (B) OPEN	STALL VLV # OPEN	

QUESTIONS

1. The servo tab on the left aileron:

A. Aids the aileron in moving to a desired position
B. Prevents the aileron from moving to a selected position too rapidly
C. Functions as a trim tab
D. A and C

2. The elevator trim tabs can be actuated:

A Mechanically
B. Electrically
C. Mechanically and electrically
D. By servo action of the geared tabs

3. The purpose of the rudder bias system is:

A. To boost rudder operation in turbulence
B. To automatically counteract asymmetrical thrust due to engine failure
C. To actuate the rudder trim tab in the event an engine fails
D. None of the above

4. If takeoff is attempted with the aileron/elevator gust lock engaged:

A. The gust lock automatically disengages.
B. A warning horn sounds.
C. Neither throttle can be advanced for full power.
D. Only one throttle can be advanced to full power.

5. The guard is removed from a pitch trim switch:

A. When trim cannot be accomplished using both elements of the switch
B. During testing, to assure that trimming can be accomplished with one element of the switch
C. The guard must not be removed under any circumstance.
D. During testing, to ascertain that trimming cannot be accomplished with one element of the switch

6. The flight control surfaces that are hydraulically actuated are:

A. All primary flight control surfaces
B. Flaps and airbrakes
C. Flaps only
D. Airbrakes only

7. Asymmetrical flap operation is prevented by:

A. A synchronizing cable arrangement that shuts off the flap unit cutoff valve, stopping the flaps
B. A synchronizing cable arrangement that mechanically connects both flap segments, forcing them to move in unison
C. A flow divider in the flap control unit
D. Nothing prevents asymmetrical operation; it can occur.

8. The purpose of lift dump is:

A. To decrease lift in flight
B. To increase drag and decrease lift on the ground
C. To increase drag in flight
D. None of the above

9. Stall warning is provided by:

A. The stick pusher
B. The stick pusher and stick shaker
C. The stick shaker
D. Airplane buffet

10. Stall identification is provided by:

A. The stick shaker
B. Airplane buffet
C. The stick shaker and stick pusher
D. The stick pusher

11. In the stall warning and identification system:

A. Flap position is integrated with angle-of-attack signals to determine critical angle of attack of the wing.
B. Flap position is not considered in the determination of critical angle of attack.
C. Flap position only is considered in the determination of critical angle of attack.
D. None of the above

12. Inadvertent stick pusher operation can be prevented by inhibiting:

A. Channel 1
B. Channel 2
C. Channel 3
D. Both channel 1 and channel 2 at the same time

13. The RUDDER BIAS annunciator goes out when:

A. Both systems are inoperative.
B. Both systems are operative.
C. Both RUDDER BIAS switches are on.
D. Both RUDDER BIAS switches are off.

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AVIONICS

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CHAPTER 16
AVIONICS

INTRODUCTION

The Hawker 800 XP avionics covered in this chapter includes the pitot-static system and the integrated flight control system. Chapter 16 describes the standard avionics package available for installation. The user should consult applicable AFM supplements and vendor manuals for additional information and information on specific systems not included in this chapter.

GENERAL

The pitot-static system includes two pitot heads, four static vent plates, two wing stall vents, and No. 1 and No. 2 air data computers.

The standard automatic flight control system includes the air data system, the autopilot and flight director, the electronic flight instrument system, the attitude heading reference system, and the standby instruments. A conventional flight control system without EFIS is optional but is not covered in this chapter. With either system, the flight director function can be used independently of the autopilot with the pilot hand-flying the airplane to satisfy the flight director commands, or the autopilot may be coupled to automatically steer the airplane to satisfy the flight director commands. The yaw damper system operates independently of the autopilot and may be engaged with or without the autopilot engaged.

PITOT-STATIC SYSTEM

GENERAL

The pitot-static system serves the computers of the air data system, standby altimeter, standby airspeed indicator, cabin pressurization system, and stall warning and identification system. Provision is made for the addition of optional equipment. Pitot supplies to the stall detectors and additional equipment can be isolated by manual selection.

Two pitot heads provide impact air pressure, four static vent plates provide static air pressure, and two wing vents provide stall vent air pressure. Water traps and drains are provided in the pipelines. The pitot heads and forward static vent plates are heated to prevent icing. The forward static plates are heated in the air only. The pitot-static and stall vent systems are shown in Figure 16-1.

PITOT HEADS

The pitot heads are mounted on each side of the fuselage nose (Figure 16-2). They provide independent supplies as follows:

- Left pitot head (P1)—No. 1 air data computer
- Right pitot head (P2)—No. 2 air data computer, standby airspeed indicator, stall detectors, and additional equipment, if connected

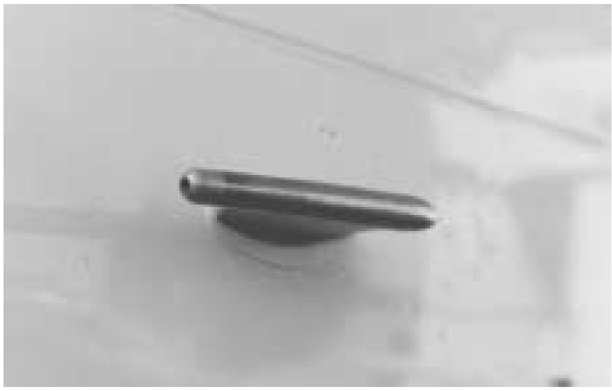

Figure 16-2. Pitot Head

Both pitot heads are electrically heated. Pitot heat is controlled by two PITOT/VANE HEAT L and R switches on the roof panel in the ICE PROTECTION group (see Chapter 10, “Ice and Rain Protection,” for additional information). Water traps and drains are provided in the pipelines.

STATIC VENT PLATES

Front and rear static vent plates are mounted on each side of the fuselage nose (Figure 16-3). Two static vents are provided in each front plate and two in each rear plate. The vents, numbered for reference, are interconnected as shown in Figure 16-1. Four static air pressure sources are thus provided to supply the following:

- Static 5—No. 1 air data computer
- Static 6—No. 2 air data computer and stall detectors
- Static 8—Standby altimeter and standby airspeed indicator
- Static 9—Cabin differential pressure indicator and optional equipment, when connected

The front plates are electrically heated when airborne only and controlled by the right PITOT/VANE HEAT switch.

Figure 16-3. Static Vent Plate

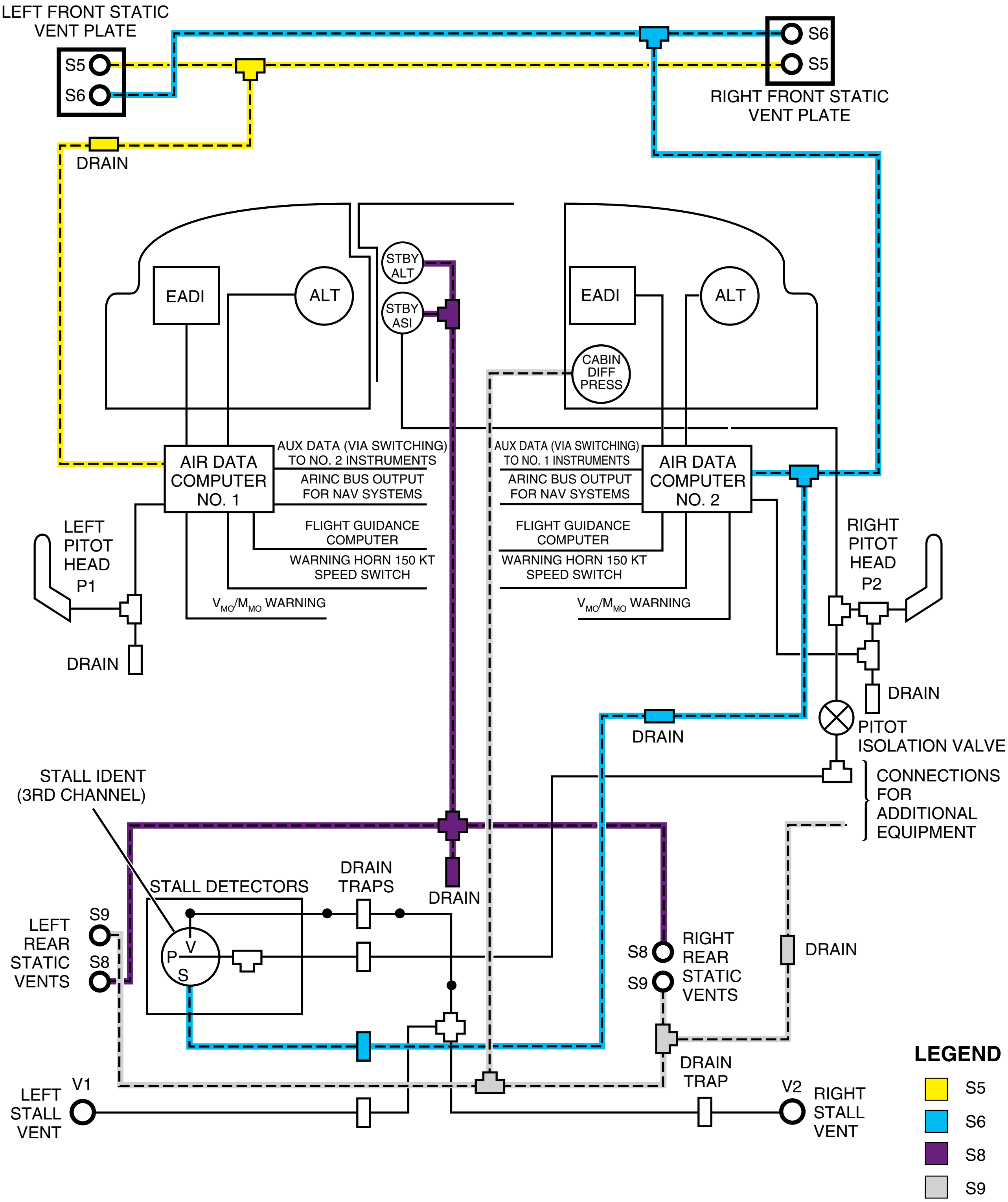

Figure 16-1. Pitot-Static and Stall Vent Systems

EQUIPMENT ISOLATION

A pitot isolation valve is located in the copilot's side console under a hinged cover (Figure 16-4). It provides a means of isolating ancillary equipment.

With the valve in the NORMAL position, pitot pressure is supplied to all relevant instruments and equipment. Depressing and turning the valve to isolate the ancillaries maintains P2 supplies to No. 2 air data computer and the standby airspeed indicator, but isolates the stall detector and additional equipment.

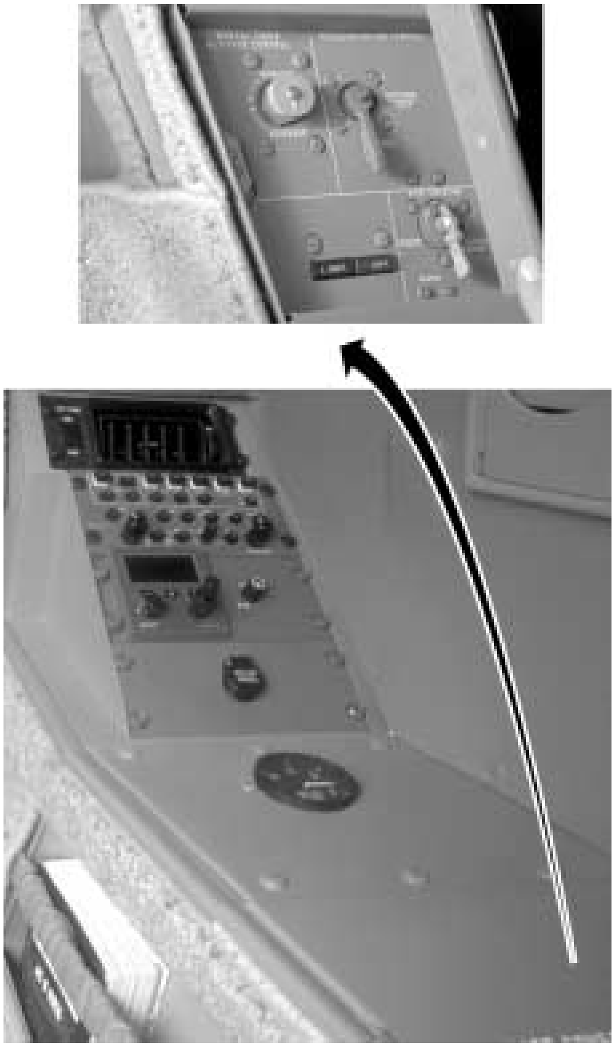

Figure 16-4. Pitot Isolation Valve

NOTE

If a discrepancy occurs between left and right airspeed indications, select PITOT ISOLATION valve to ISOLATE.

If the discrepancy is resolved, a fault exists in the ancillary equipment piping, which may cause the autopilot to become disengaged and the IDNT 3 and MWS STALL IDENT annunciators to illuminate. In this event, press either IDNT 3 annunciator to inhibit No. 3 stall identification channel. The INHIB annunciators illuminate.

If the discrepancy persists without an AIR DATA FAIL warning, the main pitot system is at fault. Continue flight, but exercise extreme caution.

STALL VENTS

A stall vent is located on the underside of each wing (Figure 16-5). The stall vents are interconnected by a pipeline which is connected to two stall detectors. One stall detector operates in the third channel of the stall warning and identification system.

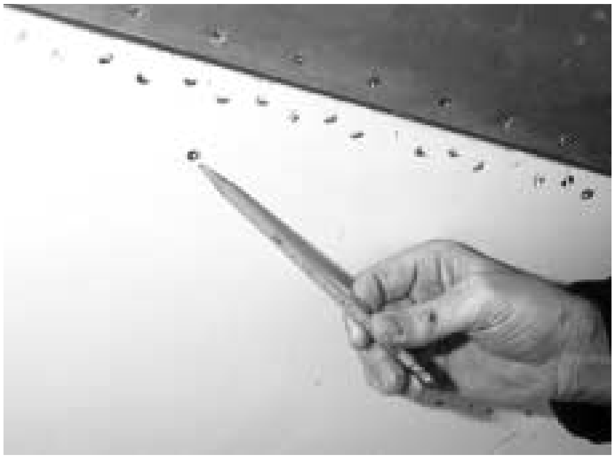

Figure 16-5. Wing Stall Vent

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CHAPTER 16A
AVIONICS

INTRODUCTION

This Chapter gives a general description of the Honeywell SPZ-8000 Digital Automatic Flight Control System (DAFCS) and its related subsystems, including the Honeywell FMZ series Flight Management System (FMS). The Honeywell pilots operating handbooks or AFM supplements have much more detailed descriptions.

GENERAL

The SPZ 8000 DAFCS is a complete, automatic flight control system providing fail operational execution of flt director guidance, autopilot, yaw damper and auto trim functions. Subsystems that make up the SPZ-8000 are:

- Dual DFZ-800 Flight Guidance System
- Dual AZ-600 Attitude Heading Reference System (AHRS) or optional dual IRS (Laser Ref II) or a third backup IRS
- Dual ADZ-810 Air Data Systems
- Dual EDZ-817/818 Electronic Flight Inst. Systems, including symbol generators
- MDZ-817/818 multifunction display systems, including symbol generators
- Dual FMZ 800 Flight Management Systems (FMS)
- Dual AA-300 Radio Altimeter Systems
- Primus Weather Radar System

- Lightning Sensor System
- TCAS
- Primus Integrated Radio System

A central serial communications network provides the ability of the subsystems to communicate with each other (Figure 16A-1). This network is called the avionics standard communications bus (ASCB). In addition to ASCB communication between subsystems there are also private line paths provided for specific, sensitive data for which fault isolation is required. These private line paths are:

- AHRS/IRS attitude & heading infomation to EFIS
- AZ-810 air data airspeed/mach & Vert. speed infomation to EFIS and to AHRS/IRS
- AZ-810 Air Data Altitude infomation to Baro altitude indicator (altimeter)
- GC-810 controller to FZ-800 guidance computer
- DC-810 display controller to SG-817/818 symbol generator
- DC-810 display controller to MG-817/818 MFD symbol generator
- SG-817/818 symbol generator to ED-800 EFIS displays
- MC-800 MFD controller to MG 817/818 MFD symbol generator
- MG-817/818 MFD symbol generator to ED-800 MFD

HONEYWELL DFZ-800 DIGITAL AUTOMATIC FLIGHT GUIDANCE SYSTEMS

DESCRIPTION AND OPERATION

The DFZ-800 Automatic Flight Control System used in the Hawker-800 XP airplane is a digital, computer based system, providing flight director, yaw damper, and autopilot functions. The system is designed to be fail-operational for the first failure for any system component with the exception of the autopilot actuators. A single actuator is used in each axis due to the high reliability of these devices.

The system uses two flight guidance computers to provide the fail-operational capability. Should a flight guidance computer fail, the second computer continues to control the airplane. A failure of the second computer is fail-passive and will not disturb the airplane.

The AFCS is comprised of the following components:

Component	Qty
1. FZ-800 Flight Guidance Computer	2
2. GC-810 Control Panel	1
3. SM-200A Aileron Servo	1
4. SM-200A Elevator Servo	1
5. SM-200A Rudder Servo	1
6. TM-420 Mach Trim Servo	1

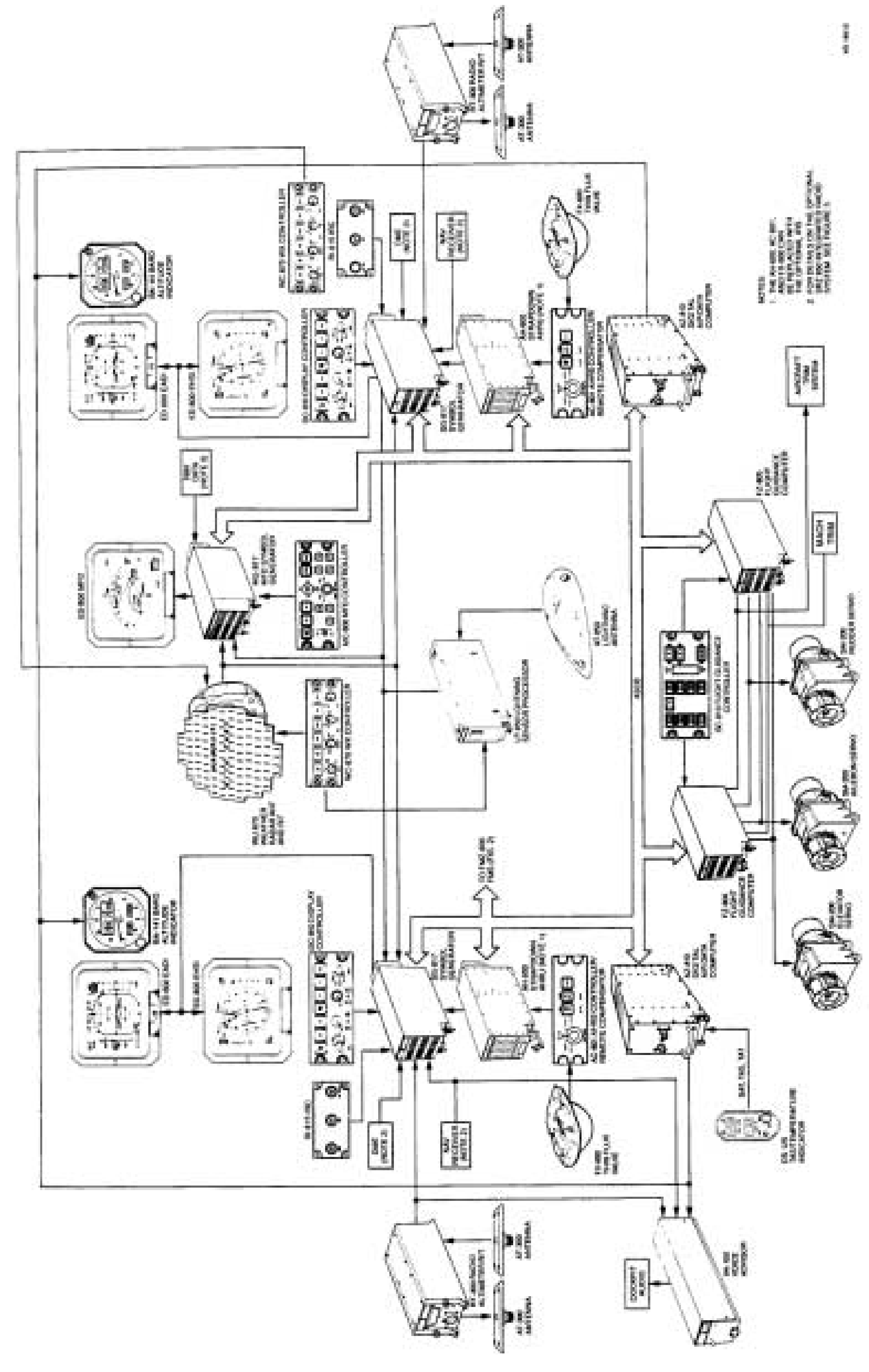


Figure 16A-1. SPZ-800 System Flow Diagram

The DFZ-800 receives information from the dual Attitude and Heading Reference Systems (AHRS), the dual Digital Air Data Computers (DADC), as well as the navigation radios (PRIMUS II). The system operates in conjunction with the Electronic Flight Instrument System (EFIS).

The interconnection between the flight guidance computers, the air data computers and the AHRS is provided by the Sperry bidirectional digital bus referred to as the Avionics Digital Communications Bus (ASCB). This dual bus is controlled by the flight guidance computer. Failure of a flight guidance computer or the bus controller function within a flight guidance computer will cause the bus control to be automatically transferred to the remaining computer.

Interconnection of primary air data information is provided by dedicated analog and ARINC 429 outputs which are independent of the ASCB.

The communications between the EFIS and the other sub-systems is carried by the ASCB. Primary data from AHRS and DADC is supplied to EFIS on dedicated digital buses.

SYSTEM CONTROLS

YD Engagement and Disengagement

The yaw damper is engaged by depressing the YD pushbutton on the GC-810 Guidance Controller. When the yaw damper is engaged, pushing the YD button on the guidance controller or the AP disconnect button will cause the yaw damper to disengage. The yaw damper is automatically engaged when the autopilot is engaged.

Autopilot Engagement and Disengagement

The autopilot is engaged by use of the AP pushbutton on the GC-810 autopilot controller. The autopilot may be disengaged by one of several methods:

a. Pressing the AP or YD pushbutton on the AFCS Controller.

b. Pressing the Go Around button located on control wheels.

c. Pressing the AP disconnect switch located on the outboard side of each of the control wheels.

d. Pressing the Electric Trim switch on the control wheels.

NOTE

The autopilot cannot be engaged on the ground.

Mach Trim

TheTM-420 Mach trim is always engaged. The pilot has no control of the mach trim. Mach trim system failure is annunciated on the main warning panel by a MACH TRIM FAIL annunciator.

HSI Select

The autopilot can be coupled to the information displayed on either HSI. The HSI SEL pushbutton on the GC-810 AFCS controller is used to select either the pilot's or copilot's HSI/DADC. The arrow annunciator illuminated on either side of the HSI SEL button and the green ---> or <--- on the EADI's show which HSI/DADC has been selected.

The flight director command bars and mode annunciations are displayed on both EADIs, automatically. Both flight directors reflect the selected HSI's data.

Navigation Source Selection

The source of the navigation information to be displayed on the HSI's is selected via the V/L, and FMS pushbuttons on the DC-810 display controller. The selected navigation source is annunciated on the EHSI.

For the selected HSI, any of four navigation sources may be selected. Pressing the V/L pushbutton selects the primary VOR/LOC navigation receiver for that side of the cockpit. The second push of the V/L pushbutton selects the VOR/LOC navigation receiver from the opposite side of the cockpit. Pressing the FMS pushbutton selects the primary Flight Management System for that side of the cockpit. The second push of the FMS pushbutton selects the FMS from the opposite side of the cockpit in a manner similar to the V/L pushbutton.When FMS is selected for display on the HSI and the NAV receiver is tuned to a localizer, the first push of the V/L pushbutton activates the localizer preview mode. This will allow arming approach while the LNAV mode is active.

Flight Director Mode Selection

The path modes of the AFCS are selected by means of the eight mode pushbuttons on the GC-810 controller (Figure 16A-2). The buttons are identified as:

HDG BNK FLC
NAV VS
APP VNAV
BC ALT

Operation of these modes is described later in this chapter. In addition to the above modes a Go Around mode is selectable using the Control Wheel mounted GA switches.

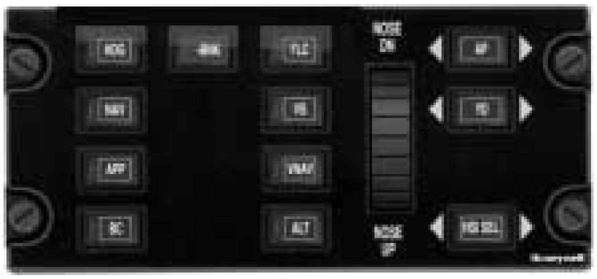


Figure 16A-2. GC-810 Controller

Pitch Wheel

The pitch wheel is used to command pitch changes. When the autopilot is in the basic pitch hold mode, the pitch wheel is used to directly change pitch attitude. The amount of pitch change is proportional to the movement of the pitch wheel.

When the VS HOLD mode is engaged, the pitch wheel is used to dial in a new reference which is displayed on the EADI's as part of the mode message.

Touch Control Steering (TCS)

The TCS button is used to uncouple the autopilot servos from the airplane without disengaging the autopilot. When the TCS button is released the autopilot will hold the current pitch and roll attitude of the airplane. If an air data mode is engaged such as FLC, VS or ALT hold, the TCS button can be used to synchronize the AFCS command to a selected reference.

Flight Guidance Channel Select

The AFCS utilizes two separate flight guidance channels, one in each of the two flight guidance computers. Under normal operation the left or number 1 channel is in control. The right or number 2 channel can be selected as the priority channel by depressing either"R AFCS" pushbuttons located next to the top right corner of the EADI's. Depressing the L AFCS pushbutton will transfer priority back to the left side computer. This selection can be made with the AP/YD engaged or disengaged.

LOCATION UPPER INSTRUMENT PANEL POWER P & CP SIDES		
LAFCS	R AFCS	FMS SX
L FAIL	R FAIL	1 2

MODES AND ANNUNCIATION

Each of the autopilot/flight director modes is described in brief below along with the annunciation which accompanies the mode. operation of the mode is the same for autopilot engaged and flight director only, unless otherwise stated.

Heading Hold/Roll Hold

The basic mode of the autopilot is heading hold/wings level. If no other mode is selected via the GC-810 the system will default to Heading Hold. If the autopilot is engaged while the airplane is in a bank, the system will engage into Wings Level and once the bank angle is less than 3 degrees for 10 seconds, the Heading Hold Mode will be engaged. The Heading Hold mode is not annunciated on the EADI as it is the default mode when no other mode is annunciated.

The autopilot will hold the desired roll attitude when the TCS button is depressed and released with the roll attitude greater than 6 degrees and less than 35. The selected bank angle will be held until the TCS button is used to lower the roll angle to less than 6 degrees. If the TCS button is released at roll angles greater than 35 degrees, the AFCS will cause the airplane to roll 35 degrees. The TCS mode causes the display of “TCS” on the EADI. This extinguishes the AP engage arrow on the GC-810 controller indicating that the autopilot servos are not engaged.

When the autopilot is engaged this basic mode is annunciated by a green flashing "ROL" for 5 seconds.

Pitch Hold

The basic pitch mode of the AFCS is pitch hold. When no flight director modes are engaged the default mode of the autopilot is pitch hold. If a roll flight director mode has been selected and no other pitch mode is selected, the flight director will engage into the pitch hold mode and a pitch command bar will be displayed.

When the autopilot is engaged, this basic mode is annunciated by a green flashing “PIT” for 5 seconds.

The reference pitch attitude may be changed via the pitch wheel or use of the TCS mode.

Heading Select

When the heading select mode is engaged, the AFCS will control the airplane through roll to follow the heading bug on the selected HSI. This mode is annunciated as “HDG” on the EADI.

The HDG Select mode may be cancelled by depressing the HDG pushbutton.

This node is limited to +/-180 degree turns. The low bank limit can be selected by depressing the -BNK pushbutton on the Guidance Controller; which is annunciated by a white low bank symbol on the top of the attitude sphere on the EADI. The low bank limit is automatically selected when climbing through 29,000 feet with bank angle less than 6 degrees.

VOR Mode

This mode is selected when the navigation receiver is tuned to a VOR frequency, the VOR nav source has been selected for display, the NAV pushbutton is depressed, and the beam deviation is below the capture threshold of 175 millivolts (2 dots equals 150 millivolts). If Heading Select is not engaged at the time the VOR mode is selected, the Heading Select Mode is automatically selected when the VOR mode is armed. Prior to capture, the VOR armed mode is annunciated as a white “VOR” message on the EADI. When the capture criteria are met, the white armed message is replaced by the green “VOR”. At this time, the HDG select mode will cancel and the green “VOR” is flashed for 5 seconds to emphasize the transition.

Passage over the VOR station will cause the system to revert to the over station submode. This causes the AFCS to ignore the erratic VOR deviation signals which are a result of the station passage. Once the system is in the over station submode, the "VOR" message is flashed for 5 seconds. When the system comes out of the over station submode, the system is again in a capture mode to allow a capture of a new outbound course if selected. The VOR mode may be cancelled by depressing the NAV pushbutton.

VAPP (VOR Approach) Mode

This mode is entered when the navigation receiver is tuned to a VOR frequency , the VOR nav source has been selected for display, the APP pushbutton is pushed, and the beam deviation is below the capture threshold of approximately 175 millivolts. operation of this mode is identical to the VOR mode with the exception that the "VAPP" message is displayed.

The VAPP mode may be cancelled by depressing the APP pushbutton.

Localizer

This mode is entered when the navigation receiver is tuned to a localizer frequency, the LOC nav source has been selected for display, the NAV pushbutton is depressed, and the beam deviation is below the capture threshold of approximately 175 millivolts (two dots equals 150 millivolts). If Heading Select mode is not engaged at the time that NAV is selected, the Heading Select mode will automatically be engaged. Prior to capture the "LOC" mode is annunciated in white on the EADI. Upon capture the LOC mode is annunciated in green and the white message is cleared. The green "LOC" is flashed for 5 seconds to emphasize the transition. The LOC mode may .be cancelled by depressing the NAV pushbutton.

Glideslope

This mode is entered when the LOC nav source has been selected for display, a localizer frequency has been tuned on the nav receiver, the APP pushbutton is depressed, and beam deviation is below the capture threshold. The capture threshold varies as a function of the rate of closure onto beam center. The faster the closure rate, the farther out the capture point. Selection of APP automatically selects the LOC mode. Prior to capture, the armed "GS" mode is annunciated in white. Upon capture the white message is cleared and the "GS" mode is annunciated in green. The green "GS" is flashed for 5 seconds to emphasize the transition. The GS mode may be cancelled by depressing the NAV or APP pushbutton.

DUAL HSI — When both the LOC and GS are on track and the radio altitude is less than 1200 feet and both nav receivers are tuned to the same ILS and valid, the AFCS will transition to the Dual HSI mode. When this mode is active, both HSI SEL arrows are illuminated and the green —> is replaced by < — > on the EADI's. In this mode both flight guidance channels are using information from both navigation receivers. This allows the approach to be continued in the event of a failure of one nav receiver. Should one receiver fail, the arrow associated with that receiver will extinguish and the mode will remain active.

LOC PREVIEW — While flying LNAV mode; with the navigation receiver tuned to a localizer frequency; the first push of the V/L pushbutton on the EFIS controller will activate the LOC PREVIEW mode. At this time pushing the APP pushbutton will arm the LOC and GS modes. The LNAV mode will remain active until the localizer deviation is within the capture threshold. At this time the EHSI will revert to the normal presentation of the short range navigation information and the LNAV mode will cancel.

BC (Back Course Localizer)

Operation of this mode is identical to that of the LOC mode with the exception that the BC pushbutton and "BC" mode messages are used. The BC mode does not allow use of a simultaneous GS mode.

The BC mode may be cancelled by depressing the BC pushbutton or by selecting the NAV (LOC) mode.

Go Around

Selection of Go Around mode is made via the switch located on the control wheels. Selection of this mode will disengage the autopilot. The flight director will command a twelve degree pitch up attitude and wings level. The "GA" mode is annunciated on the EADI. The Heading Select mode may be selected while in go around to replace the wings level command.

The Go Around mode may be cancelled by selecting another pitch mode, engaging the autopilot or by depressing the TCS pushbutton.

LNAV (Lateral Navigation)

This mode is enabled when the FMS nav source has been selected, a lateral valid is received, and the NAV pushbutton is depressed. Selection of this mode without a valid signal will result in a white "LNV" armed message. Upon receipt of the LNAV valid the AFCS will clear the armed message and will display a green "LNV" active message. The green "LNV" is flashed for 5 seconds to emphasize this transition. The LNAV mode may be cancelled by depressing the NAV pushbutton.

FLC (Flight Level Change)

This mode is enabled when the following conditions are met:

- The desired altitude is selected.
- The FLC pushbutton is depressed.
- A maneuver towards the selected altitude is initiated.

The mode annunciation is a green "FLC" on the EADI. The mode automatically synchronizes to the current airspeed (below 29,000 ft) or MACH number (above 29,000 ft). The airspeed reference is displayed in blue on the bottom left corner of the EADI. The reference may be changed by use of the "IAS/MACH" knob on the remote instrument controller (RI-815) . once a new airspeed is selected, the AFCS will command a smooth transition to acquire the new reference which is also displayed by a solid blue bug on the airspeed scale.

The reference may also be changed by using the TCS button. If the airspeed is changed while in the FLC mode and the TCS button is depressed, the reference airspeed will synchronize to the current airspeed at the time the TCS button is released. The airspeed to which the AFCS is commanding is always displayed on the EADI.

The IAS reference automatically switches to MACH reference above .63M in a climb. The MACH reference automatically switches to IAS above 285 kts in a descent. The reference may be manually switched between IAS and MACH at any time by depressing the "PUSH CHG" pushbutton on the RI-815.

The AFCS will not fly to an airspeed reference outside the normal aircraft flight envelope. The AFCS limits the commanded airspeed to the maximum allowable speed of the aircraft minus a small built in offset. This is annunciated by an amber "MAX SPD" on the EADI.

The FLC mode may be cancelled by the pitch wheel or selecting another vertical mode.

VS (Vertical Speed Hold

This mode is enabled when the VS pushbutton is depressed. The annunciation is a green "VS" on the EADI. The reference vertical speed can be changed by use of the pitch wheel. The vertical speed displayed changes as the pitch wheel is moved. once a new vertical speed is selected, the AFCS will command a smooth transition to acquire the new reference which is displayed in blue on the bottom right corner of the EADI.

The reference vertical speed can also be changed by use of the TCS button. Operation is similar to that of the FLC mode. The vertical speed to which the AFCS is commanding is always displayed in the "----FPM" window on the EADI.

Selection of VS will cancel ALT, FLC, GA, VNAV and GS. The VS mode can be cancelled by depressing the VS pushbutton or by selecting another vertical mode.

As in FLC mode, the AFCS will not fly to an airspeed reference outside the normal aircraft flight envelope while the VS mode is active.

ALT (Altitude Hold Mode)

This mode is enabled by depressing the ALT pushbutton. It mav also be entered by using the ALT SEL mode described below. A green “ALT” message is displayed while altitude hold is active. The reference altitude may be changed by using the TCS to maneuver to a new altitude and then releasing the TCS button. Using the pitch wheel cancels the ALT mode. Selecting the ALT mode will cancel the IAS, VS, ALT SEL, VNAV, GA and GS modes. The ALT mode may be cancelled by depressing the ALT pushbutton , by selecting another vertical mode or use of the pitch wheel

ALT SEL (Altitude Preselect)

This mode is enabled by selecting the desired altitude through the copilot's RI-815 and initiating a maneuver towards the selected altitude. The ALT SEL armed mode is automatically annunciated by turning the blue "ASEL" in the altitude preselect window white. The aircraft can be maneuvered toward the selected altitude using pitch,VS,VNAV, or FLC. The AFCS will transition to the ALT capture mode at a point away from the selected altitude which is dependent upon the vertical speed. The ALT capture mode is annunciated by the green flashing "ASEL" message. The green ASEL is flashed for 5 seconds to emphasize this transition. When the aircraft has flared onto the new altitude, the system will transition to ALT hold and the "ASEL" will be replaced by the “ALT.”

Note

> Changing the selected altitude while in the capture phase will cause the active ALT SEL mode to be cancelled and the ALT SEL mode to arm. This is indicated by turning the blue ASEL white.
>
> With the Autopilot not engaged (FD only) there should be a lateral or a vertical mode selected for the ALT SEL mode to arm automatically.

VNAV (Vertical Navigation Mode)

With the FMS selected for display on the HSI selected side, activation of the VNAV pushbutton will select the Vertical Navigation mode and overrides all active pitch flightdirector modes. In the VNAV mode AFCS tracks the vertical flight profile of the FMS with following submodes possible:

VFLC (Flight Level Change)

VFLC will operate identically to FLC except that the target speed and the altitude from the FMS are used for climb or descent. VFLC will also engage if VALT is engaged and FMS initiates a climb or descent. A third possible way to enter VFLC mode is when VALT or VPTH arm is engaged and the FLC button on the GC-810 is pushed. The FMS FLC mode is Annunciated on the EADI by a green “VFLC”at the vertical capture location. The target speed is displayed on the airspeed scale by a magenta bug.

VASL (Altitude Capture)

VASL will operate identically to ALT SEL mode. VASL will arm as soon as VFLC or VPTH is engaged. The FMS ALT SEL arm mode is also annunciated by turning the blue "ASEL" in the altitude preselect window white.

When the mode captures, “VASL” in green will be displayed on the EADI vertical capture location. The mode annunciation will flash for 5 seconds emphasizing the transition from arm to capture.

VALT (Altitude Hold)

VALT will operate identically to ALT mode. VALT will engage automatically after VASL has captured the target altitude. VALT will also engage whenever VNAV pushbutton is pushed and the aircraft is within 250 feet of the FMS target altitude. The FMS ALT mode is annunciated on the EADI by a green "VALT" at the vertical capture location.

VPTH (Vertical Path Mode)

VPTH mode is used to fly a fixed flight path angle to a vertical waypoint during descent. VPTH mode will engage whenever FMS initiates a path descent which may occur while in VFLC or VALT modes. When the mode captures, "VPTH" in green is annunciated on the EADI at the vertical capture location. The mode annunciation will flash for 5 seconds to emphasize this transition.

MONITORS

AFCS Failure Monitors

The AFCS incorporates monitors which assess the validity of the computer and other system components. These monitors are in addition to those monitors which provide fail-passive and failoperational operation described below. The power supply, the micro-processors, and other hardware within the system are monitored for availability. If a failure is detected the system is rendered inoperative.

Upon application of electrical power to the system the AFCS goes through a power-up pre-flight test. The “L FAIL” annunciator is on during the test. Upon successful completion of the test by both computers, the “L FAIL” annunciator turns off and the “L AFCS” annunciator turns on. If a failure is detected in either flight guidance computer the appropriate “L (R) FAIL” annunciator latches on.

A failure within the flight guidance computer which prevents operation of the flight director function will result in a red “FD” flag on the EADI.

AFCS Performance Monitors

The AFCS, also, incorporates monitors within hardware and software that ensure the safety of the system. These monitors provide the fail-passive and fail-operational capability of the system. It is beyond the scope of this document to describe these monitors in detail, but the following list identifies the monitor functions and the resultant annunciation.

Servo Monitors

The servo control loops are monitored to detect failures of the servo amplifiers, loss of feedback, or existence of an uncommanded output. If the monitors detect a failure condition within the priority computer the computer will be shut down and the remaining computer will continue operation. Assuming that the left computer is active (engaged), the amber “L FAIL” annunciator turns on. This is followed by a green “R AFCS” annunciation to indicate that the right side computer is now active.

NOTE

> In the event of a nuisance monitor trip within the priority computer, the control will be transferred to the remaining computer and the down computer will perform an automatic reset. If the down computer is successful in resetting itself, no fail annunciation will be displayed. The down computer will not try to reset itself if the AFCS is in the capture or track phase of LOC, BC, VAPP or GA.

Should a failure occur that affects both computers, e.g. an open winding in the servo motor, the second computer will detect the failure and the AFCS will disengage. The "AP DISC" annunciator will turn on and the autopilot disengage horn will sound for one second.

When both channels of the AFCS are operating normally, failures of the pitch, roll, or yaw servo controls cause the AFCS to transfer from the priority computer (channel) to the standby computer and display a "L (R) FAIL" annunciation. When only one computer is operational, failures in the pitch and roll servo controls cause the autopilot to disengage. Failures in the yaw damper servo control cause both the autopilot and the yaw damper to disengage.

Failures of the servo controls prior to engagement will result in a FAIL annunciation when engagement is attempted.

Closure Monitors

The internal signals within the AFCS are monitored in order to determine that the system is correcting to remove errors. For example, the roll attitude, roll rate, and servo rotation must all be in the proper direction to correct for any roll axis errors or the roll monitors will trip. The DISENGAGE and FAIL annunciations are the same as mentioned in the Servo Monitors above.

Mistrim monitors

Roll

The aileron servo current is monitored in order to detect an out-of-trim condition in which there is a steady state load on the aileron servo. When a steady state load of significant magnitude exists for 20 seconds the amber "AIL TRIM L(R)" annunciator will turn on.

Pitch

The elevator servo is monitored to detect a long term load on the servo. When this condition is detected the "ELEV TRIM UP (DN)" annunciator will turn on. This message may appear momentarily during normal operation for maneuvers which require large trim changes.

Sensor Data Monitors

Flagged Failures

AHRS/DADC

Both channels of the AFCS receive data from both AHRS and from both DADC's. When both AHRS are valid the AFCS uses data from both AHRS to compute the aircraft control functions. The DADC data is handled in much the same manner with the exception that only one DADC provides reference data for the VS hold, FLC, ALT hold and ALT SEL functions. The DADC being used follows the HSI SEL arrows. Failure of one AHRS or one DADC will not affect the AP or YD engage status. Failure of one DADC will disengage any air data mode if the failed DADC is on the side indicated by the HSI SEI. arrow. That is, if the HSI SEL arrow is pointing to the pilot's side, the failure of the pilot's DADC will cause the air data modes to disengage. Failure of the copilot's DADC will have no effect on the mode status.

Navigation Receiver Failures

VOR Mode

Loss of valid from the navigation receiver for more than 30 seconds will cause the VOR modes to disengage. While the aircraft is passing over a station the loss of the nav valid will not cause the mode to disengage because the radio information is considered unreliable and is not used by the autopilot.

APP (LOC & GS), BC and VOR APP modes

Loss of the navigation receiver valid for more than 4 seconds will cause the mode to disengage.

When both HSI SEL arrows are illuminated (indicating that both receivers are being used simultaneously) a loss of the nav receiver valid will not disengage the mode. The AFCS will continue to fly the approach using the remaining good receiver. A loss of the second receiver will cause the mode to disengage.

Unflagged Failures

AHRS

Upon loss of valid data from both AHRS or upon a disagreement (split) between the data from the two AHRS, the autopilot and yaw damper will disengage.

It is possible to have valid AHRS data displayed on the primary instruments and not have valid data to the AFCS since separate data paths are used. Dual AHRS data failures will cancel all Flight Director modes.

Digital Air Data Computer (DADC)

Upon loss of valid data from both DADC's or upon a disagreement (split) in the data from the two DADC's, the AFCS will disengage. It is possible to have valid altitude and altitude alerter functions on the panel instruments and not have valid data to the AFCS since separate data paths are used. Dual DADC data failures will cancel VS, FLC, ALT, and ALT SEL flight director modes.

Comparator Monitors

AHRS

If the pitch or roll attitude from the two AHRS differ by 6 degrees and both AHRS are valid, the AFCS will cancel the FD mode and disengage the AP and YD.

DADC

The data from the two DADC's are compared in the same manner as the AHRS data. Unflagged differences between the data cause the AP and YD to disengage. The flight director modes are retained.

GS and LOC Deviation

The two navigation receivers are compared in the AFCS during Dual HSI Approach. If the localizer or glideslope do not compare, the AFCS selects a receiver based upon the magnitude and polarity of the two signals. The HSI SEL arrow associated with that failed receiver will be extinguished, identifying that the system is no longer in the DUAL HSI mode. If the two receivers once again agree, the system will again use both receivers and the DUAL HSI mode will again become active.

AP AND YD ENGAGE STATUS

Any change in the status of the engagement of either the yaw damper or the autopilot will be annunciated. if either the AP or YD pushbuttons are depressed, the system will illuminate the appropriate arrow on the GC-810. If the AP or YD are disengaged due to a fault in the AFCS the “AP DISC” disengage lights will stay on. A single push of the AP disconnect button on the control wheel will reset the lights.

MODE ANNUNCIATION

All AFCS mode annunciation is displayed on the EADI's directly in front of each pilot. Armed modes are annunciated in white active modes are annunciated in green. Transition from an armed mode to an active mode is emphasized by flashing the green annunciation for 5 seconds.

OPERATING LIMITS

MODE	ROLL ANGLE LIMIT	ROLL RATE LIMIT
HDG SEL	+/- 27 degrees +/- 17 deg. (low bank)	4 degrees/sec
NAV		
VOR Capture	+/- 24 degrees	4 degrees/sec
VOR Track	+/- 24 degrees	4 degrees/sec
LOC Capture	+/- 30 degrees	7.5 degrees/sec
LOC Track	+/- 24 degrees	5.5 degrees/sec
VOR APP Cap	+/- 30 degrees	7.5 degrees/sec
VOR APP Track-	+/- 24 degrees	4 degrees/sec
* LNAV	+/- 30 degrees	5.5 degrees/sec
BC Capture	+/- 30 degrees	7.5 degrees/sec
BC Track	+/- 27 degrees	5.5 degrees/sec
HDG HOLD	+/- 13 degrees	5.5 degrees/sec
ROLL HOLD	+/- 35 degrees	5.5 degrees/sec

MODE	PITCH LIMIT	PITCH G LIMIT
ALT HOLD	+/-20 degrees	.1g
ALT SEL	+/-20 degrees	.05g
GS	+10, -15 degrees	.1g
FLC	+/-20 degrees	.3g
VS HOLD	+/-20 degrees	.1g
GO AROUND	+12 degrees	FD Only
PITCH HOLD	+/-20 degrees	.3g

* Lateral Navigation source may have lower limits

EMERGENCY PROCEDURES

Emergency

In the event of Autopilot malfunction:

The SPZ-8000 system is a fully monitored autopilot and will disengage without a significant attitude or heading change if a malfunction should occur.

The pilot may disengage the autopilot by:

1. Press (momentarily) the AP/YD disengage button on the pilot's or co-pilot's control yoke.
2. Press (momentarily) the AP or YD button on the Guidance Controller.
3. Open the Autopilot circuit breakers.

In the event of an engine failure:

1. Disengage the autopilot/yaw damper, retrim the airplane and re-engage the Yaw Damper and Autopilot as desired.
2. Before any significant power changes during single engine operation, disengage the Autopilot and Yaw Damper, retrim the aircraft as required and re-engage the Autopilot and Yaw Damper as desired.

Maximum Altitude Losses during malfunction test were:

Altitude loss was not appreciable during malfunctions during all phases of flight, including climb, cruise, descent and coupled approaches.

ATTITUDE & HEADING REFERENCE SYSTEM (AHZ-600)

DESCRIPTION AND OPERATION

The dual AHRS installation is approved as the primary attitude and heading reference source. Several operational modes are provided for continued availability of attitude and heading in the event of certain system failures. The standard system consists of dual attitude and heading reference units, dual AHRS controllers and dual flux valves (Figure 16A-3, Figure 16A-4, and Figure 16A-5).

Pitch, roll and heading are provided to the Electronic Flight Instrument System (EFIS) and the Automatic Flight Control System (AFCS). Attitude and heading data is dupplied to other aircraft systems such as the weather radar antenna, RMI's and flight control system.

The AHZ-600 AHRS is an all attitude inertial sensor system that differs from conventional vertical directional gyro systems in that the gyroscopic elements are rate gyros which are strapped down to the principal aircraft axes. It has no gimbals. The spinning mass follows the airframe and its output signals are rate sensitive. A digital computer contained in the AHZ-600 mathematically integrates the rate data to obtain heading, pitch and roll. A flux valve and three accelerometers provide long term references for the system.

The “DG” mode disables the automatic slaving of the heading outputs. Entry into this mode can only be achieved by momentarily pressing the HDG/DG pushbutton on the AHRS controller. Entry into the DG mode occurs when the pushbutton is released and is confirmed by the DG 1 (DG 2) annunciation on the EHSI display. AHRS operation in the “DG” mode results in a heading system which is similar to a free directional gyro, and which

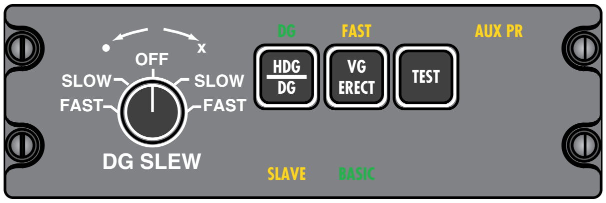


Figure 16A-3. AHRS Controller

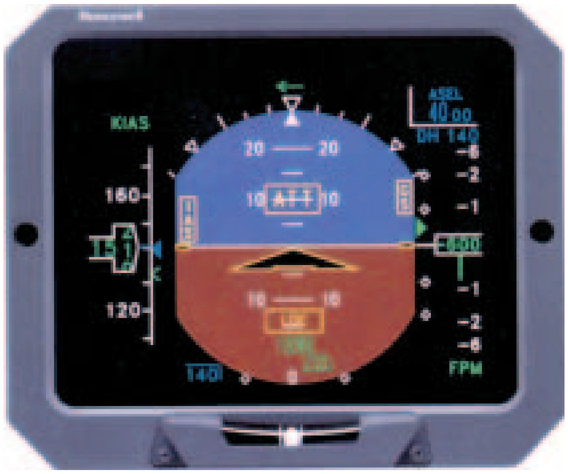


Figure 16A-4. ADI with "ATT" Comparator

is subject to drift and turn error. For this reason, AHRS operation in the "DG" mode results in reduced heading accuracy.

While in the "DG" mode, the heading card may be manually set to any heading using the DG SLEW knob on the AHRS controller. The control is inactive in the "Slaved" mode. Two slew speeds are provided for each direction. The SLOW slew position is located at the mechanical detent and is used for fine heading card adjustments. The FAST slew position is located at the mechanical limits of the control knob and is used for large heading card adjustments. The knob will automatically center to the OFF position when released.

Turning the knob clockwise causes the heading card to indicate increasing headings. The HDG flag on the EHSI will come into view initiating a slew command and will clear upon releasing the control knob.

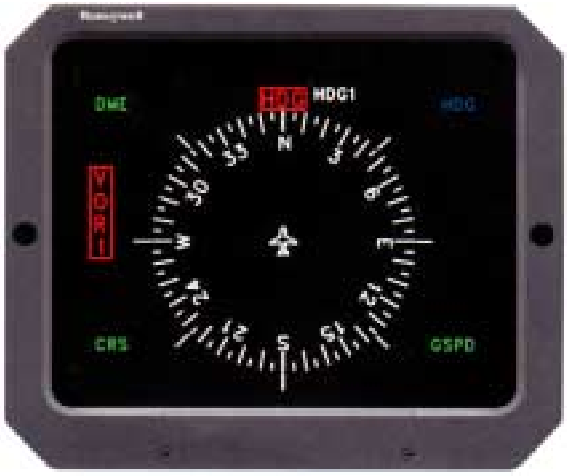


Figure 16A-5. HSI with "HDG" Flag

In the "Slaved" mode, a loss of valid flux valve data will result in a HDG flag on the heading instruments (EHSI and RMI) and the illumination of the red SLAVE annunciator on the AHRS controller. The HDG flag will clear upon subsequent entry to the DG mode, but the SLAVE annunciator will remain lit (Figure 16A-5).

Upon exit from the "HDG" mode, the AHRS performs an automatic synchronization of the heading outputs to the present flux valve magnetic heading. This feature may also be used if a heading error should develop, while in the "Slaved" mode. The error may be quickly removed by momentarily entering the "IDG" mode and returning to the "Slaved" mode. This is performed by pressing the HDG/DG pushbutton on the AHRS controller twice.

In the "Slaved" mode the difference between the indicated heading and the flux valve heading is displayed on the Slave Error Indicator (Heading Sync Indicator) located on the EHSI. The card has two symbols: a cross ("+") and a dot ("o"). The cross ("+") and dot ("o") are displayed in a fixed position with a moveable pointer on the EHSI.

During straight and level flight the Slave Error Indicator will generally be centered with excursions toward the cross or dot occurring over a 20 to 30 second time period.

This activity is normal and indicative of good magnetic heading data. In turns, the display may show a steady dot or cross. Following return to straight and level flight, the indicator will return to the "centered" condition within 2 minutes.

In the DG mode, without a SLAVE fail indication on the AHRS controller, the system may be manually cleared using the DG SLEW knob and the Slave Error Indicator. This is performed by turning the SLEW control knob in the appropriate direction to cause the Slave Error Indicator to center between the cross and dot.

The verticality of the AHRS may be checked or corrected during unaccelerated flight by activating and holding the VG ERECT pushbutton on the AHRS controller for a minimum of 10 seconds. The ATT flag will be displayed on the EADI while the VG/ERECT pushbutton is held. FAST annunciator on the AHRS controller will illuminate to indicate entry into the "FAST ERECT" mode of operation. Upon releasing the VG/ERECT pushbutton, the ATT flag will clear and roll attitude will become active, and the FAST annunciator will extinguish.

System Test

The AHRS is equipped with automatic self test which is performed when power is first applied. The test lasts 5 seconds and provides the following visual outputs on the EFIS:

- 10 degrees pitch up
- 20 degrees right wing down
- North heading, turning at 3 degrees per second toward East
- BASIC annunciator ON
- ATT flag Valid for 2.5 seconds, then invalid
- HDG flag Valid for 2.5 seconds, then invalid.
- Rate of Turn Indicator standard rate (3 deg/sec) right turn

NOTE

> When EFIS is powered up simultaneously with AHRS, the self test on AHRS will not be visible due to the warm up time of the display heads.

The flags will remain invalid until initialization is complete.

The AHRS system requires approximately 3 minutes to initialize following application of power. The initialization is complete when the ATT and HDG flags clear on the ADI and HSI. During the initialization, the aircraft must remain stationary. Wind gusts and aircraft buffeting are not limiting in this respect. Similarly, all normal preflight operations, including engine starts and passenger loading, may be carried out while the AHRS is initializing. If the initialization requires more than 3 minutes, the AHRS may have detected excessive aircraft motion. If aircraft movement has occurred during the initialization, the AHRS will recycle and a new initialization will commence.

NORMAL OPERATION

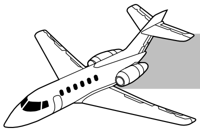
Ground Initialization

The aircraft shall remain stationary on the ground until the attitude, and heading, flags are pulled out of view. Normal passenger and cargo loading, engine start, and engine run-up procedures may be performed during the initialization. Wind buffeting is not limiting in this respect. Taxiing or towing the aircraft during AHRS iniation is prohibited.

Check that 3 minutes following power connection to the DC buses (#1 AHRS P.E. and #2 AHRS PS-2) the attitude, and heading flags are out of view.

If the flags do not pull after 5 minutes, the AHRS is not serviceable. This time may be greater at higher latitudes.

Prior to flight, verify proper AHRS and display functions by observing the AHRS test sequence. Proper display movement, flag



operation, and "BASIC" lamp operation shall be observed. These tests are performed automatically upon power application.

Normal pre-flight taxi checks of pitch, roll, heading, and, rate of turn, shall be made on each system.

If the aircraft is moved during AHRS initialization, both AHRS will re-initialize; hence, the power up may take longer than 3 minutes.

NOTE

> If operating outside region 1, and at (N or S) latitudes, certain combinations of turns, immediately followed by a large sustained aircraft acceleration can result in ten degrees (maximum of 25 degrees) of heading error. This error will be removed in approximately two to five minutes after the acceleration is completed.

ABNORMAL OPERATION

Takeoff In Basic Mode

Take-off with one AHRS in "Basic" mode is not allowed.

NOTE

> If a third attitude source, i.e., IRS, is available, it can be used as the primary data in place of the AHRS in Basic mode.

Takeoff In DG Mode

Take-off with one AHRS in the "DG" mode is not recommended.

NOTE

> If a third heading source, i.e., IRS is available, it can be used as the primary data in place of the AHRS in DG mode.

Flight Operations In Basic Mode

If a malfunction causes the AHRS to revert to the "Basic" mode, it will be annunciated on the AHRS controller. Normal flight operations may be continued in the "Basic" mode subject to the limitations of the Flight Manual.

Following entry into the "Basic" mode, the pilot shall avoid sustained, shallow banked turns of less than 6 degrees (e.g. a constant turn to hold DME arc). In addition, particular attention shall be paid to ensure correct aircraft trim.

If an attitude error should develop, it may quickly be removed by activating the VG ERECT switch for 10 seconds while in unaccelerated flight.

Flight Operations In DG Mode

If a Heading Flag is observed during a flight, the "DG" mode may be selected by momentarily pressing the HDG DG pushbutton.

The AHRS heading shall be checked every five minutes with reference to a known accurate heading source. Any errors may be removed by using the DG SLEW knob on the AHRS controller to set the heading card to agree with the known reference.

Attention shall be paid to the "DG" mode heading when flying NDB approaches to ensure correct heading data.

In-Air Initialization

In-air initialization is not recommended.

NOTE

> If Laseref II is used for attitude and heading information instead of AHRS, please reference YG1782B Inertial Reference System Pilot's Manual Document No. 95-8351.

Failure Modes and Reversion

On side AHRS attitude and heading information is compared with a cross side attitude and heading. An amber "ATT" annciator is displayed when there is a 6° miscompare in pitch or roll with no red failure warnings annunciated (Figure 16A-4).

An amber "HDG" annunciator is displayed when there is a 6° miscompare in headings with no red failure warnings annunciated (Figure 16A-7).

Activation of either amber "ATT" or "HDG" comparator causes it to flash for 10 sec. and then stay on steady. If either of these amber comparators comes on, a determination must be made as to which side attitude or heading information is in error. When this is determined, cross side AHRS information can be selected with the AHRS reversion switchlight.

AHRS 1 OVHT	SG		AHRS		ADC		EFIS OVHT	EFIS FAN FAIL
	Norm	Rev	Norm	Rev	Norm	Rev		

Figure 16A-6. Reversionary Switch Panel

Failure of attitude or heading information is annunciated by a respective red "ATT" or "HDG" light in the EADI or EHSI (Figure 16A-5). Complete failure of the respective AHRS unit would be annunciated by; in the EADI

a) removing the pitch scale and roll pointer

b) removing the FD bars

c) removing sky/ground differentiation (All blue display)

d) displaying a red boxed "ATT" flag in the EHSI

a) removing TRU/MAG annunciator

b) removing the HDG bug

c) removing the course pointer and deviation bar

d) removing any map information

e) removing to/from display

f) the heading card goes to north up

g) displaying a red boxed "HDG" flag

Once again, selection of the failed side "AHRS" reversionary switchlight will bring cross side AHRS information to the failed side.

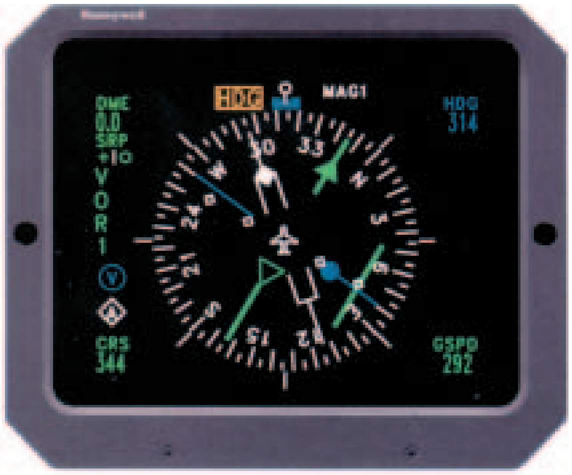
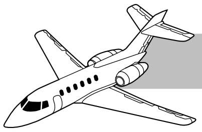


Figure 16A-7. ATT Comparator



ADZ-810 AIR DATA SYSTEMS

DESCRIPTION AND OPERATION

The AZ-810 receives pneumatic information from the aircraft's pitotstatic system and transforms these pressures to digital words of Ps and Pt. The total air temperature probe is used as the input for calculation of Total Air Temperature. Baro correction is brought in from a potentiometer on the BA-141 as a resistance ratio proportional to the displayed baro setting. The actual baro correction of altitude is performed in the AZ-810. These air data inputs are then used to calculate all other air data parameters. In addition, a thorough monitoring system continuously evaluates the internal operation of the AZ-810 and controls the hardware discrete valids as well as the valid bits on digital bus outputs.

Air data parameters that are calculated include Indicated Airspeed, Mach Number, Pressure Altitude, Baro Correction, Baro Corrected Altitude, Vertical Speed, Total Air Temperature, Static Air Temperature, True Airspeed, Static Source Error Correction, Digitalized Pressure Altitude (for mode C transponder), V_{MO}, and an overspeed warning discrete for the aural warning. Private line bus outputs are provided for the BA-141, and transponder. The AZ-810 also provides air data parameters to the Sperry Flight Control System via the Avionics Standard Communication Bus (ASCB). Dual ARINC 429 outputs are provided to supply information to other aircraft systems, such as TAS to the area navigation systems, and air data to the cabin display.

All altitude alerting computation, display control, and output driver are contained in the AZ-810. The set knob on the altitude alert knob provides a signal input to the AZ-810 which is used to slew the Altitude Alert word to a new value. The baro corrected altitude calculated by the AZ-810 is continuously compared to the Altitude Alert word to provide alert light and aural warning outputs. The Altitude Alert digital word is also provided to the Flight Control System via the ASCB to be used in Altitude Preselect operation.

The AZ-810 Digital Air Data Computer has an operating range of -1000 to +60,000 feet of altitude, and 30 to 450 knots of indicated airspeed. The Mach number calculation range is from 0.10 to 1.00.

The left air data computer normally supplies airspeed/mach, vertical speed, and altitude to the LH flight instruments. The right air data computer normally supplies airspeed, vertical speed and altitude to the RH flight instruments.

SYSTEM TEST

Air Data Computer Test

Honeywell Phase III

1. AC and DC bus bars energized.
2. Radio and EFIS master switches selected on.
   a. Verify air data systems valid.
3. On autopilot control panel verify HSI selected left. On overhead roof panel “CG,” press and hold #1 ADC test switch. Check that:
   a. Pilot's altimeter momentarily flags then drives to 1,000 feet.
   b. VSI runs to 5,000 ft/min climb.
   c. Airspeed runs to 325 knots.
   d. Altitude select runs to 12,000 feet.
   e. TAS indicator shows dashes.
   f. ADC FAIL annunciator illuminates.

4. While holding #1 ADC test switch, press and hold #1 V_{MO}/M_{MO} test switch.
    - a. A pulsing overspeed warning horn will be heard.
    - b. Release test switches.
5. On autopilot control panel select HSI select switch right. On overhead roof panel "CG," press and hold #2 ADC test switch.
    - a. Verify same results as step 3 on ~copilot's instruments.
6. Repeat step 4 with #2 ADC test and #2 V_{MO}/M_{MO} test switches with the same result.

NOTE

The low V_{MO} test switches and the ADC test switches on the overhead panel have the same results.

With fuel in the ventral tank an overspeed warning will result when ADC or low V_{MO} test switches are pressed. Also, airspeed will display 290 knots.

7. Select copilot's ADC transfer switch.
    - a. Observe ADC #1 annunciation on pilot's and copilot's EADI.
    - b. Observe REV annunciation on copilot's transfer switch.
    - 8. Select pilot's ADC transfer switch.
    - a. Observe copilot's transfer returns to normal.
    - b. ADC #2 annunciation on pilot's and copilot's EADI.
    - c. Observe REV annunciation on pilot's transfer switch.
9. Deselect items 7 and 8 and observe the removal of the ADC #1 or ADC #2 annunciator.

NOTE

During engine start, ADC power is transferred to the #4 battery. It is normal for a momentary flag to occur on airspeed, vertical speed, and altimeter at the end of the start sequence when ADC power transfers back to the normal power source.

Failure Modes and Reversion

On side airspeed is compared to cross side airspeed. An AMBER "IAS" comparator comes on in the EADI when there is a 5 knot difference for more than 2 seconds. As long as both speeds are valid and at least one side is 90 kts or above, the amber "IAS" comparator initially,flashes for 10 sec. then stays on steady until speeds once again agree (Figure 16A-8).

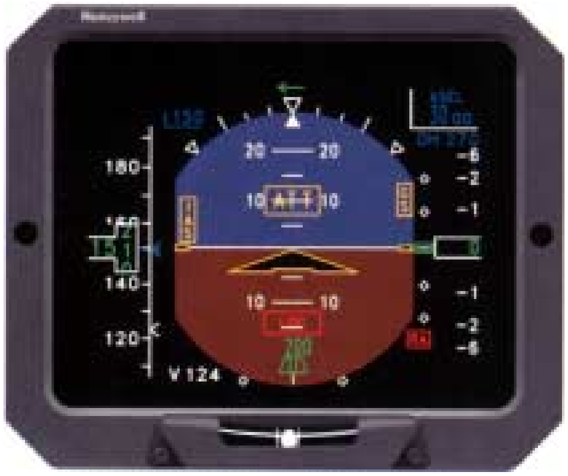


Figure 16A-8. Amber IAS Comp

Failure of an individual component such as airspeed, or vertical speed or the altimeter would be shown as:

a) in the case of airspeed, the airspeed scale & digital readout disappearing, replaced by a RED "IAS" (Figure 16A-9).

b) in the case of vertical speed, the vertical speed scale are digital readout replaced by a RED "VS" (Figure 16A-9).

c) in the case of the altimeter, a red off failure flag covers the digital portion of the altimeter face.

d) in the case of the SAT/TAS (if installed) guage, blank displays replaced by a single dash in the middle of each display.

Selecting "ADC" switchlight on the reversionary panel (Figure 16A-6) may or may not get valid info back, depending on whether it is an individual instrument problem or a partial ADC failure.

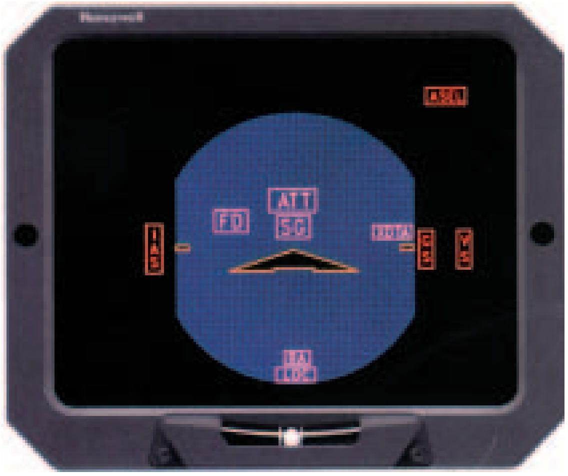


Figure 16A-9. Red "IAS" Readout

Complete failure of the respective air data computer would be shown by a RED "IAS" and no airspeed scale or digital readout, a red "VS" and no vertical speed scale or digital readout and a red off flag covering the digital portion of the altimeter. All three indications would occur simultaneously (Figure 16A-9).

To allow the opposite ADC to power the failed sides instruments, select ADC switchlight on the reversionary panel (Figure 16A-6). The green "Normal" light will be replaced by an amber "REV" light. Valid airspeed and vertical speed should now replace the red "IAS" and "VS" flags in the EADI. In addition, the failed side EADI will display an amber "ADC 1" or "ADC 2". The sending side EADI will display a green "ADC 1" or "ADC 2". The failed altimeter is not included in the reversion and will remain flagged. Other sources of altitude will have to be used (Figure 16A-9).

EDZ-817/818 EFIS

DESCRIPTION AND OPERATION

The Electronic Display System shown in Figure 16A-10 is comprised of three subsystems. The Pilot's Display System, the Copilot's Display System and the Multifunction/Weather Radar System. The pilot's and copilot's systems are identical, with both providing ADI and HSI information to the flight crew. The Multifunction/Weather Radar display system provides long and short range navigation maps, checklists and weather radar, and serves as a data source and back-up symbol generator for both the pilot's and copilot's display system.

All five electronic displays are identical and interchangeable. A conventional slip/skid indicator is attached to the top display to be used in conjunction with the attitude presentation. The electronic displays use a combination of manual and photoelectric dimming to enable clear readability under varying light conditions.

The pilot's and copilot's system contain identical symbol generators which are the focal point of information flow in the systems. The symbol generator's major task is to convert the information present at its inputs to video and deflection formats required by the display units to form the ADI and HSI displays. The MFD symbol generator provides the navigation maps and checklists for display on MFD display, and also serves as a backup symbol generator for either side.

The pilot's and copilot's display system both contain a display controller and a source controller. The display controller provides the means by which the pilot can control the display formatting, such as full or partial compass display or single cue or cross pointer display. The display controller is also used to select navigation and bearing sources for display. Auxiliary switches in the cockpit are used to select alternate heading and attitude sources, EFIS air data sources, and to select reversionary mode. The MFD system also contains two controllers. The MFD controller, provides for the selection and control of the MFD formats, modes, waypoint designator, and checklist. The weather radar controller selects the radar modes and adjustable quantities which may then be displayed on either the MFD or EHSI.

EDZ-817 CONTROLS

DC-810 Display Controller (One Each Side)

The Display Controller (Figure 16A-11) provides the means by which the pilot can control the EFIS display formatting. Also included on the Display Controller are the bearing select knobs. The following paragraphs describe the controller functions:

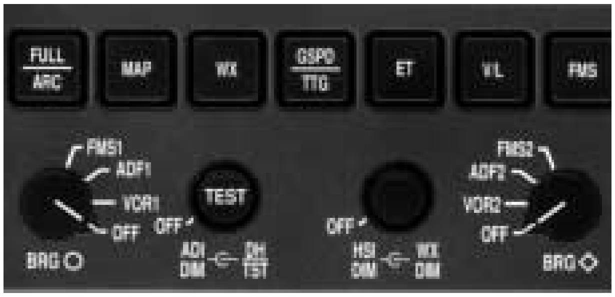


Figure 16A-11. Display Controller

FULL/ARC Button

The FULL/ARC button is used to change the EHSI display from full compass to partial compass format. In the full compass mode, 360' of heading are displayed. In the partial compass mode, 90° of heading are displayed. Successive toggling of the FULL/ARC button changes the display back-and-forth from the full compass to the partial compass display.

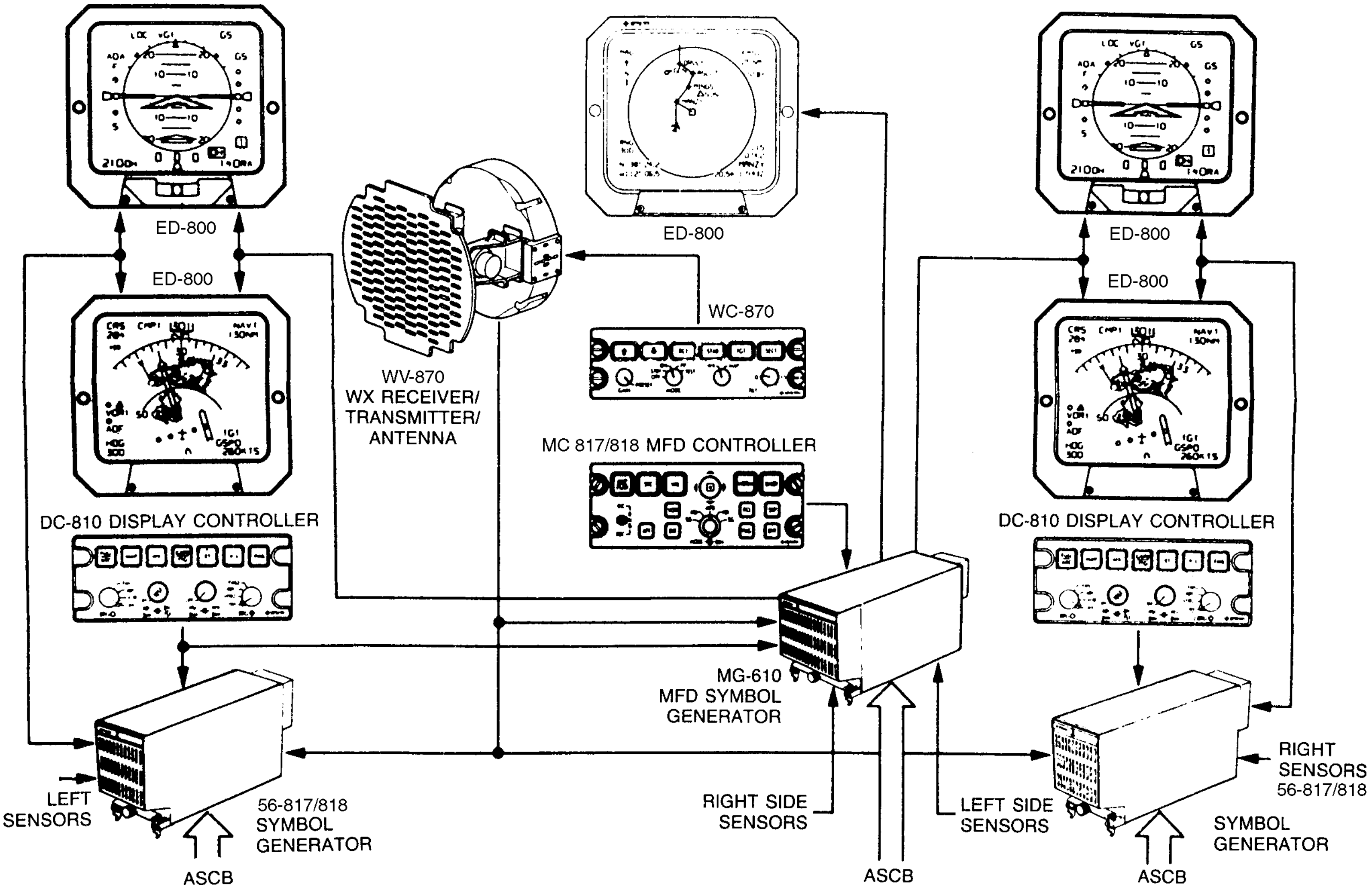

ED-800
ED-800
DC-810 DISPLAY CONTROLLER
LEFT SENSORS
56-817/818 SYMBOL GENERATOR
ASCB
WV-870 WX RECEIVER/ TRANSMITTER/ ANTENNA
ED-800
WC-870
MC 817/818 MFD CONTROLLER
MG-610 MFD SYMBOL GENERATOR
RIGHT SIDE SENSORS
ASCB
LEFT SIDE SENSORS
ED-800
ED-800
DC-810 DISPLAY CONTROLLER
RIGHT SENSORS 56-817/818
SYMBOL GENERATOR
ASCB

Figure 16A-10. EDZ-817/818 Electronic Flight Instrument System Block Diagram

MAP Button

By pressing the MAP button, the full compass display may be changed to the partial compass format allowing one waypoint for each bearing pointer and VOR/DME ground station positions to be displayed. The information will only be displayed if within the range selected. If in WX mode, activation of the MAP button will select the MAP format with weather radar information.

Weather Radar (WX) Button

The WX button is used to call up weather radar returns on the partial compass MAP display of the EHSI. If the EHSI is in the full compass MAP mode initially, pressing the WX button changes the display to the partial compass MAP mode and displays weather radar returns from the weather radar or a weather test pattern. If presently in MAP mode, activation of the WX button will super-impose weather radar information over the MAP information. A second push of the WX button will remove the weather information.

Ground Speed/Time-To-Go (GS/TTG) Button

By pressing the GS/TTG button, ground speed or time-to-go will alternately be displayed in the lower right corner of the EHSI. The power-up condition is ground speed displayed.

Elapsed Time (ET) Button

By pressing the ET button, elapsed time is displayed. ET replaces to GS/TTG display. Pressing the ET button the first time will reset the displayed time to zero. The toggling sequence of the ET button is reset, start, stop.

VHF Omni Range/Localizer (V/L) Button

By pressing the V/L button, VOR/LOC information is selected for display on the EHSI. This information is also sent to the FZ-800 Flight Guidance Computer. This button toggles between NAV 1 and NAV 2. The power-up condition is the on-side NAV source.

Flight Management System (FMS) Button

By pressing the FMS button, FMS information is selected for display on the EHSI. This button toggles between FMS 1 and FMS 2. The power-up condition is the on-side FMS.

Bearing (BRG) Source Select Knobs

The EHSI has the capability of displaying two independent bearing pointers. The selectable bearing sources for each pointer are as follows:

BRG O	BRG ◊
OFF	OFF
VOR 1	VOR 2
ADF 1	ADF 2
FMS 1	FMS 2

DIM Controls

The dimming system employed by the EDZ-817 EFIS is semi-automatic. Two inputs contribute to the overall display brightness of each ED-800 Electronic Display:

- Ambient light sensed by the photosensors on each ED-800
- Setting of the dimming controls

The DIM pot sets the nominal intensity for each display. The photosensors located on each ED-800 cause the light output of each display to be modulated about the nominal intensity as a function of the light incident on each display.

ADI DIM Control

The ADI DIM control dims the raster and stroke writing on the EADI. Turning the control to the OFF position causes the EADI tube to go blank and displays the EADI mode on the EHSI tube.

HSI DIM Control

The HSI DIM control dims both stroke and raster writing on the EHSI. Turning the control to the OFF position causes the EHSI to go blank. The EHSI display can be selected on the MFD.

WX DIM Control

The WX DIM control dims only the raster on the EHSI which contains weather radar information. Turning the control to the OFF position removes the weather radar display.

Decision Height (DH) Knob

Rotation of the inner DH knob allows the decision height displayed on the EADI to be adjusted between 20 and 990 ft in 10-foot increments. By rotating the DH knob completely counterclockwise, the decision height display may be removed from the EADI display.

Test (TST) Button

The TST button allows for testing the radio altimeter and EFIS when on the ground or the radio altimeter only when in the air, if not in the glide slope capture mode.

RI-815 Instrument Remote Controls

The following paragraphs describe each control on the pilot's and copilot's controller (Figure 16A-12).

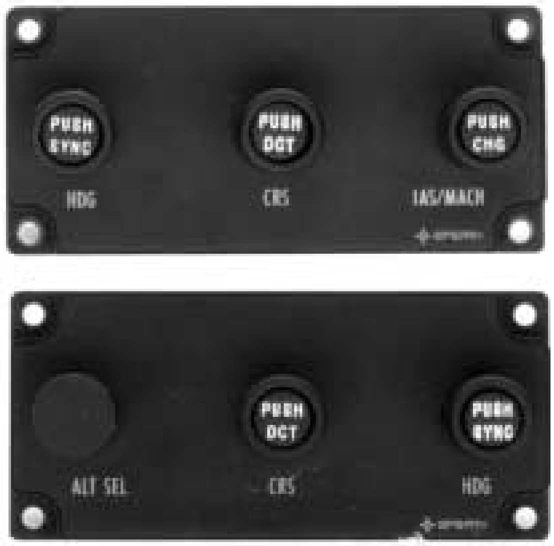


Figure 16A-12. RI-815 Instrument Remote Controllers

Heading (HDG) Select Knob and PUSH SYNC Button

The HDG knob allows selection of a heading reference by the positioning of the heading bug on the pilot's or copilot's EHSI.

Activation of the PUSH SYNC button causes synchronization of the heading bug to present heading (lubber line).

Course (CRS) Select Knob and PUSH DCT Button

The CRS knob allows selection of VOR or ILS course by the positioning of the course select pointer on the pilot's or copilot's EHSI.

The PUSH DCT button allows automatic selection of a TO direction desired VOR course having zero deviation.

When FMS is selected, the system provides the capability for the pilot to preselect an on-side preselected course arrow to be set as desired. When turning the knob, the DTK digital readout is replaced by the digital value of preselected course. The digital preselected course readout will disappear 5 sec after rotation of the CRS knob has ceased.

Pilot's IAS/MACH Knob and PUSH CHG Button

The IAS/MACH knob allows selection of EADI displayed IAS or MACH reference speed for the Flight Level Change (FLC) flight director mode. The power-up value for selected speed is 80 kts IAS.

Repeated activation of the PUSH CHG button causes alternate selection of IAS and MACH. Activation of the PUSH CHG button does not cause the IAS/MACH bug to move, but simply changes the nature of the readout. Power-up state is IAS.

Copilot's Altitude Select (ALT SEL) Knob

The ALT SEL knob controls the preselect altitude displayed on the EADI. The altitude is set in 100-foot increments.

MULTIFUNCTION DISPLAY (MFD) SYSTEMS

MDZ-817/818 Controls

General MDZ-817/818 MFD System Information

The MDZ-817/818 Multifunction Display System has three major subfunctions. One of the most significant is its ability to backup either of the two EFIS symbol generators or one of the EHSI displays. Should one SG fail, the pilot can select the MFD symbol generator to take over operation of the failed side's displays with all functions and operations unchanged. The EFIS DC-810 controller on that side will continue to operate the display formats as before. The MFD display can also be used as a backup in the event of an EHSI display failure.

The MFD system greatly expands on the navigation mapping capabilities of the EFIS. This is primarily due to the fact that the MFD display area can be used exclusively for map formats without the need for the essential heading and NAV data that the EHSI also has to contain. Some of the additional information that can be added to the traditional MAP display of waypoint locations includes waypoint, airport, and VOR identifiers and the TO waypoint time-to-go or ETA. This additional data is supplied by the Flight Management System for display on the MFD. The MFD system also has a north-up plan function in addition to the usual heading-up MAP display. Both formats make use of a designator controlled by the MC-800 controller joystick. The position of the designator can be automatically transmitted to the FMS to be used in defining a new waypoint.

MC-800 MFD Controller

The MC-800 MFD Controller provides the means by which the pilot can control the MFD display modes and format. The following paragraphs describe the controller functions (Figure 16A-13).

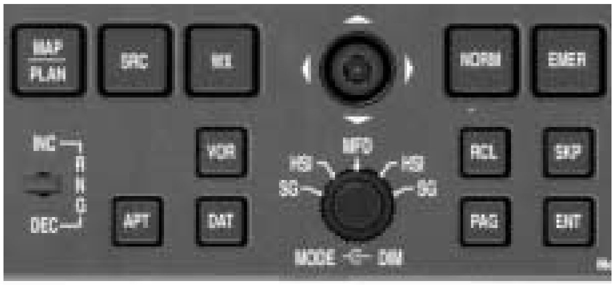


Figure 16A-13. MC-800 MFD Controller

MAP/PLAN Button

The MAP/PLAN button alternately selects the heading-up MAP display or the north-up PLAN mode for display.

Source (SRC) Button

The SRC button alternately selects the FMS 1 and FMS 2 source of long range navigation data for mapping.

Weather (WX) Button

The WX button is used to call up weather information for display on the MFD. When weather is displayed, the MAP range is controlled by the WC-870 Weather Radar Controller.

Several toggling sequences can occur depending on the weather radar configuration on the aircraft. Refer to Table 16A-1.

Normal (NORM) Button

The NORM button provides entry into the MFD's normal checklist display function. The normal checklist is arranged in the order of standard flight operations. Button actuations cause presentation of the normal checklist index page that contains the lowest order in-

Table 16A-1. WX BUTTON FUNCTIONS

WX Button	1st Push	2nd Push	3rd Push	4th Push	5th Push	6th Push
Single WC-870 Controller (no LSS)	WX/Map	WX Only	—	—	—	—
Single WC-870 Controller (with LSS)	WX/Map	WX Only	WX/LSS	—	—	—
Dual WC-870 Controller (no LSS)	Pilots WX/Map	Pilots WX Only	Copilots WX/Map	Copilots WX Only	—	—
Dual WC-870 Controller	Pilots WX/Map	Pilots WX Only	Pilots WX/LSS	Copilots WX/Map	Copilots WX Only	Copilots WX/LSS

complete and unskipped checklist with the active selection at that checklist. The SKP, RCL, PAG, and ENT buttons and the joystick provide control of this function.

Emergency (EMER) Button

The EMER button provides entry into the MFD's emergency checklist display function. Actuation of EMER button results in the presentation of the first page of the emergency checklist index, with the active selection at the first checklist.

Skip (SKP), Recall (RCL), Page(PAG), and Enter (ENT) Buttons and Joystick

These buttons (PAG for checklist only) and joystick are used to control the checklist and operation of the designator on the MAP. The following describes the checklist and designator control.

Checklist

- SKP — Actuation skips the active selection to the next item.
- RCL — Actuation results in presentation of the page containing the lowest order skipped item with active selection at that item.
- PAG — Actuation advances the page count. The active selection is the lowest order incomplete item on that page. If there are no incomplete items on the page, the active selection is the first item on the page.
- ENT — Operation is dependent upon display if:

  on an index page, actuation results in display of the checklist corresponding to the active index line selection. The checklist is presented at the page containing the lowest order incomplete item with the active selection at that item. If the checklist had previously been completed, the system forces all items in the checklist to be incomplete and presents the first page of the checklist with the active selection at the first item.

  on a checklist page, actuation forces the active selection to be completed and advances the active selection to the next incomplete item. If ENT is actuated with the active selection at the last item in a checklist, the operation depends upon the completion status of the checklist.

  If the checklist is not complete (one or more items skipped), the system presents the page containing the lowest order incomplete item with the active selection at that item.

  If the checklist is complete (all items completed), the system presents the index page containing the next higher order checklist with the active selection at that checklist.

- Joystick — The joystick provides additional paging and cursor control. Each actuation results in the action described:

  UP moves the active selection to the lower order item.

  DOWN moves the active selection to the next higher order item (this is identical to SKP).

  LEFT results in presentation of the previous page.

  RIGHT results in presentation of the next page (this is identical to PAG).

Designator Control

- SKP — Actuator skips the designator's home position to the next displayed waypoint. When actuated with the designator at the last displayed waypoint, the designator shall return to present position.

- In the plan mode display, the SKP function allows the pilot to sequence through all the waypoints of his active flight plan.

- RCL — When the designator is not at its home position, actuation of RCL recalls the designator to the home position. Actuation with the designator at its home position recalls the designator to present position (if not already there).

- ENT — When the designator is offset, actuation of ENT causes the LAT/LON of the designator to be transmitted to the selected LRN as a requested waypoint.

- Joystick — The joystick provides four direction control of the designator: up, down, left, and right. The distance and course to the designator from its home position is displayed in the lower right corner of the display.

VHF Omni Range (VOR) Button

The VOR button is used to add VOR/DME symbols and designators to the MAP and PLAN displays. The toggling sequence is:

- First button push will display the tuned VOR/DMEs.

- Second button push will display all VOR/ DMEs within display range, up to a maximum of 4.

- Third button push will remove VOR/DME symbology from the display.

Airport (APT) Button

The APT button is used to add airport symbols and designators to the MAP and PLAN displays. The toggling sequence is:

- First button push will display the two closest airports to present position from the data base.

- Second button push will display all airports within display range, up to a maximum of 4.

Data (DAT) Button

The DAT button is used to add long range navigation information to the MAP and PLAN displays.

- First actuation — The following data will be added to the lower right corner of the display.

  Waypoint identification

  Estimation Time of Arrival (ETA) in Greenwich Mean Time (GMT) at the TO waypoint if known; otherwise, Time-To-Go (TTG) to the TO waypoint

  Distance TO waypoint

  All displayed waypoint identifiers

- Second actuation — If no destination information is known, this step shall be to data OFF. However, if destination identification, ETA, or TTG is known, this step shall replace the TO waypoint data as described above with the destination data. If some destination data is known but the waypoint identification is not, the mnemonic DEST shall be used in place of the waypoint identification.

Increase (INC)/Decrease (DEC) Range (RNG) Switch

This switch increases or decreases the selected range (5, 10, 25, 50, 100, 200, 300, 600, and 1200 NM) if the weather radar (WX) mode is not selected. When WX is selected, the range is controlled by the WC-870 Controller.

Mode Selector

This is a five position rotary switch used to select the following MFD modes of operation.

- MFD — Normal MFD operation

- HSI (counterclockwise), Pilot EHSI Backup — In this position, the MFD drips its normal display and displays the pilot side EHSI. The EHSI display can be changed by using the pilot's DC-810 EFIS Display Controller.

- SG (counterclockwise), Pilot SG-817/818Backup—In this position, the MFD display will go blank and the MFD symbol generator is driving the pilot's EADI and EHSI displays. The EADI and EHSI displays are still controlled by the pilot's DC-810 EFIS Display Controller.
- HSI (clockwise), Copilot EHSI Backup—In this position, the MFD drops its normal display and displays the copilot's side EHSI. The EHSI display can be changed by using the copilot's DC-810 EFIS Display controller.
- SG (clockwise), Copilot SG-817/818 Backup —In this position, the MFD display will go blank and the MFD symbol generator is driving the copilot's EADI and EHSI. The EADI and EHSI displays are still controlled by the copilot's DC-810 EFIS Display Controller.

DIM Control

This knob controls overall MFD CRT dimming in addition to the ED-800 automatic photoelectric dimming.

MFD Weather Radar Mode

There are two basic weather radar (WX) modes. One, WX with map, is similar to the EFIS WX mode. Two range rings are presented at selected range and half range. The other basic mode is WX only. This differs from WX with MAP by presenting four range rings with azimuth lines at 0°, ±30°, and ±60°. Both modes provide standard weather formats using different colors to denote various intensity levels. Normal WX colors are black, red, yellow, green, magenta, and blue. Ground mapping utilizes black, cyan, yellow, and magenta colors. Turbulence is shown in white. All modes are annunciated on the MFD display.

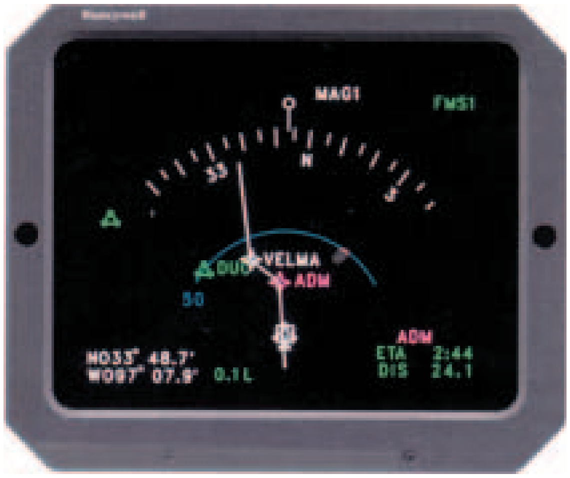


Figure 16A-14. MFD "MAP" Mode With Flight Plan Infomation

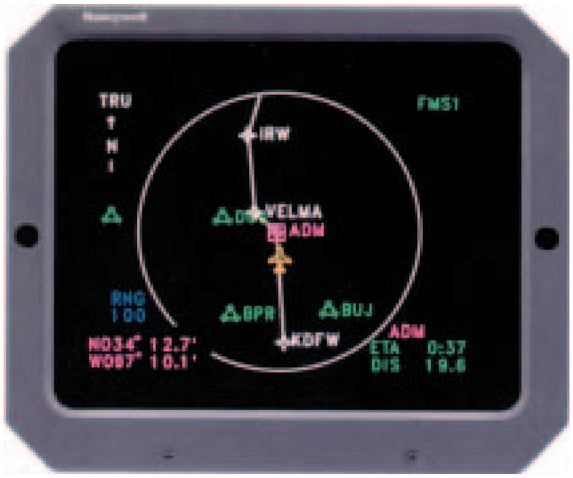


Figure 16A-15. MFD "PLAN" Mode With Flight Plan Infomation

A target alert annunciator is provided below the mode annunciator to warn of level 3 targets 7.5° on either side of the aircraft flight path. A cyan annunciates this feature, changing to an amber flashing when active. At the WX mode location, RCT is displayed in cyan characters for Rain Echo Attenuation Compensation Technique (REACT).

A weather radar failure will remove the raster weather display and force the mode annunciator to display WX in amber characters.

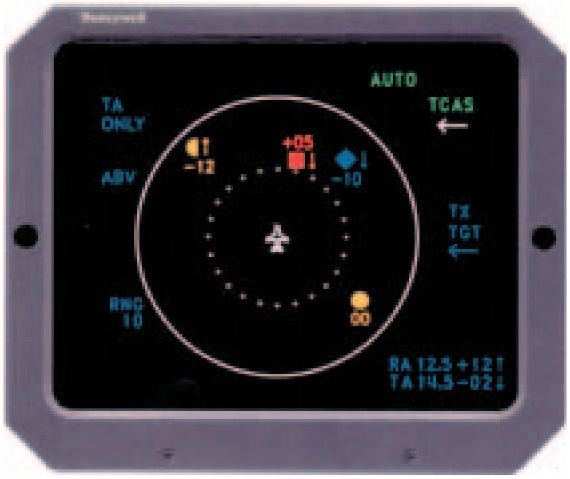


Figure 16A-16. MFD With TCAS II Display

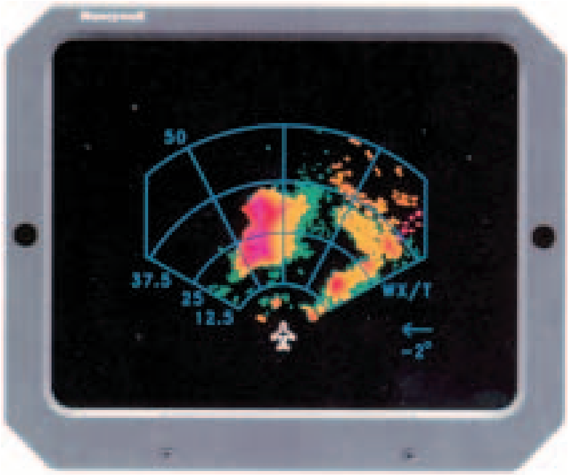
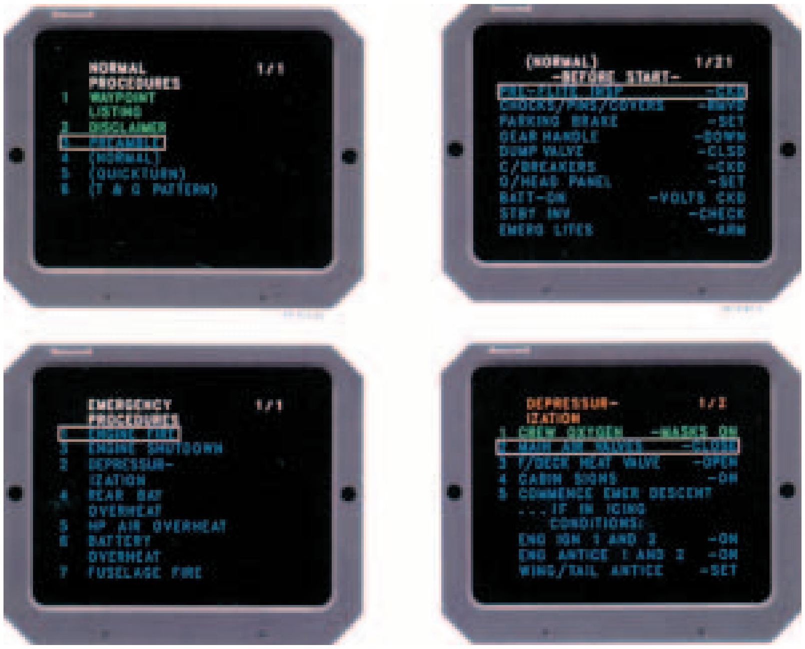


Figure 16A-17. MFD With Radar Only Display



```
NORMAL                  1/1
PROCEDURES
1   WAYPOINT
    LISTING
2   DISCLAIMER
3   PREAMBLE
4   (NORMAL)
5   (QUICKTURN)
6   (T & G PATTERN)
```

```
   (NORMAL)             1/21
     -BEFORE START-
PRE-FLITE INSP          -CKD
CHOCKS/PINS/COVERS     -RMVD
PARKING BRAKE           -SET
GEAR HANDLE            -DOWN
DUMP VALVE             -CLSD
C/BREAKERS              -CKD
O/HEAD PANEL            -SET
BATT-ON           -VOLTS CKD
STBY INV              -CHECK
EMERG LITES             -ARM
```

```
EMERGENCY               1/1
PROCEDURES
1   ENGINE FIRE
2   ENGINE SHUTDOWN
3   DEPRESSUR-
    IZATION
4   REAR BAY
    OVERHEAT
5   HP AIR OVERHEAT
6   BATTERY
    OVERHEAT
7   FUSELAGE FIRE
```

```
  DEPRESSUR-            1/2
  IZATION
1 CREW OXYGEN     -MASKS ON
2 MAIN AIR VALVES    -CLOSE
3 F/DECK HEAT VALVE   -OPEN
4 CABIN SIGNS           -ON
5 COMMENCE EMER DESCENT
  ...IF IN ICING
     CONDITIONS:
  ENG IGN 1 AND 2       -ON
  ENG ANTICE 1 AND 2    -ON
  WING/TAIL ANTICE     -SET
```

Figure 16A-18. MFD In Various Checklist Displays

A magenta TX is displayed where WX is annunciated when the P-870 is ON, and weather is not selected for MFD display.

A WX arrow will be used to show which WC-870 is the master controller. The arrow will not point toward an invalid controller.

For additional weather radar details, refer to the Honeywell Pilot's Operationg Handbook.

ELECTRONIC ATTITUDE DIRECTOR INDICATOR (EADI)

Depending on whether the aircraft is TCAS II equipped (818 symbol generators) or not (817 symbol generators) there are some minor differences in the EADI display as shown.

The EADI combines the familiar true sphere-type attitude display with lateral and vertical computed steering signals to provide the commands required to intercept and maintain a desired flight path. The EADI provides the following display information:

- Attitude Display
- Excessive Pitch Chevrons
- Flight Director Command Cue(s)
- Flight Director Mode Annunciations
- HSI Select Annunciation
- AP Engage/TCS Annunciation
- Vertical Deviation
- Localizer Deviation and Rising Runway
- Radio Altitude
- Decision Height Setting
- Decision Height Annunciation
- Marker Beacon Annunciations
- Vertical Track Alert
- IAS Scale and Readout
- IAS Acceleration Trend
- IAS/MACH Reference Bug and Readout
- Max Speed
- Speed Bug from ~FMS
- Digital Mach
- Cat 2 Window
- Cat 2 Excessive Deviation
- Reversionary Annunciations
- Flags
- Altitude Preselect Readout
- Comparison Monitors
- V-Speed Bugs
- Vertical Speed Scale and Display
- Low Bank Indication
- Vertical Speed Target Reference

ELECTRONIC HORIZONTAL SITUATION INDICATOR (EHSI)

ED-800 Used As An EHSI

The EHSI combines numerous displays to provide a map-like display of the aircraft position. The indicator displays aircraft displacement relative to VOR radials, localizer, and glide slope beam. The desired display (FULL, ARC, WX, MAP or MAP/WX) is selected with switches on the DC-810 Display Controller. The EHSI provides the following display information:

FULL, ARC, WX, MAP or MAP/WX Modes

- Heading (rotating dial) and airplane symbol
- Lubber line and MAG/TRU annunciator
- Selected Heading Bug
- Selected Heading Digital Display
- FMS Drift Bug

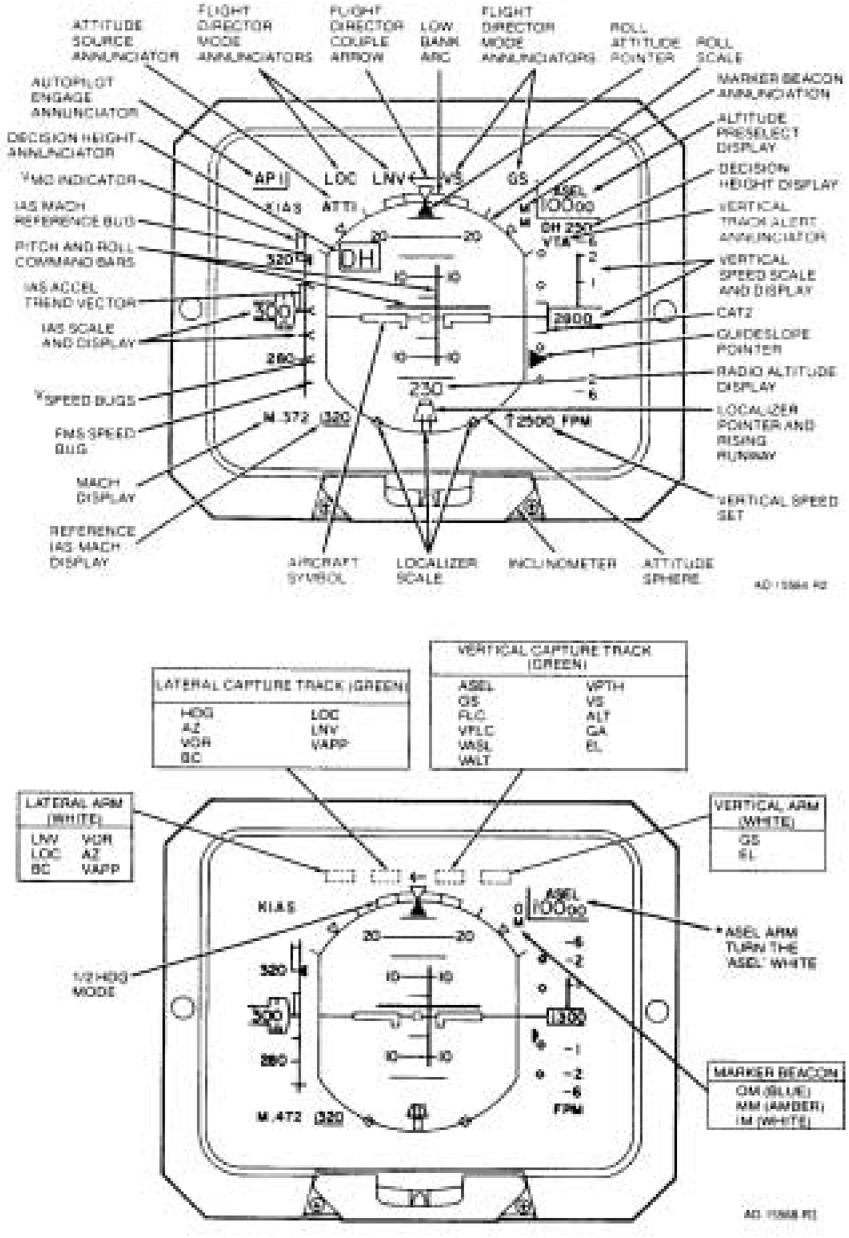
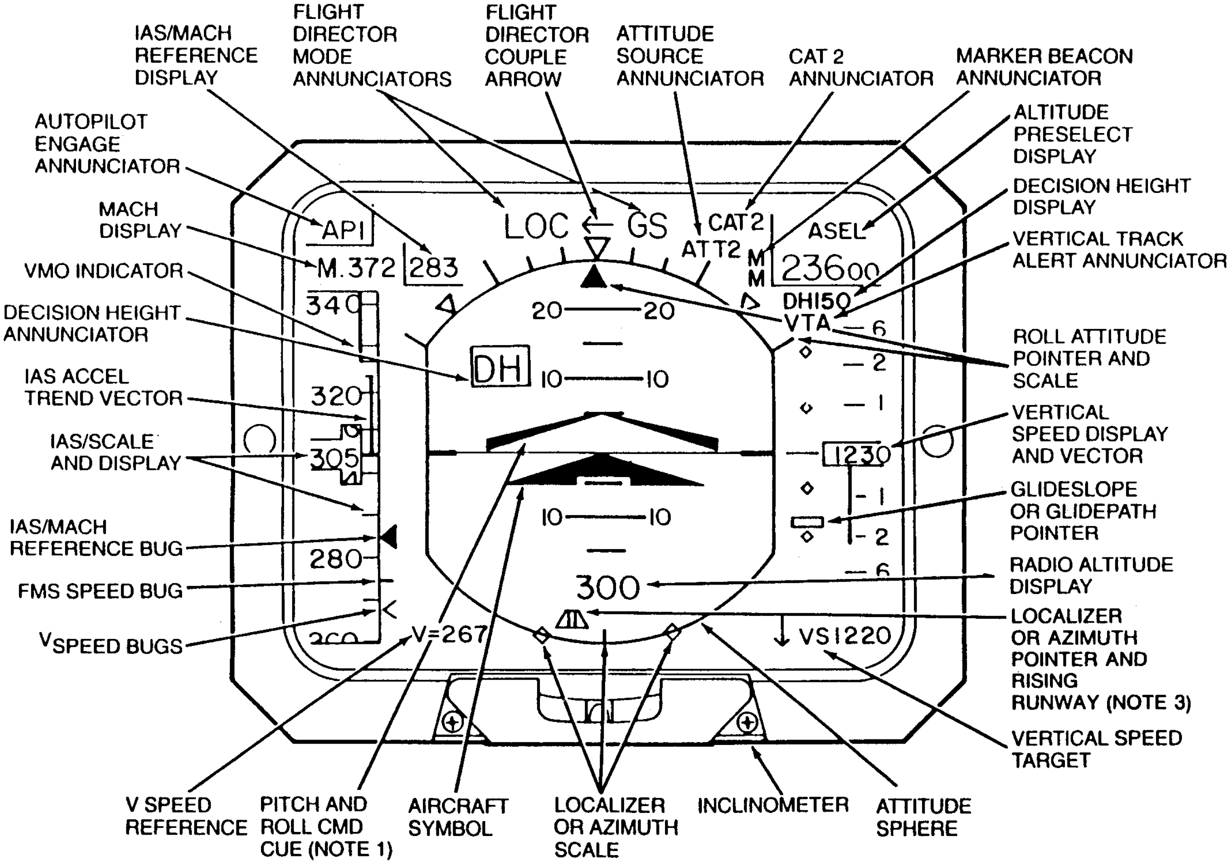


Figure 16A-19. ED-800 EADI Displays and Annunciators (Non TCAS II Aircraft)


IAS/MACH REFERENCE DISPLAY
FLIGHT DIRECTOR MODE ANNUNCIATORS
FLIGHT DIRECTOR COUPLE ARROW
ATTITUDE SOURCE ANNUNCIATOR
CAT 2 ANNUNCIATOR
MARKER BEACON ANNUNCIATOR
AUTOPILOT ENGAGE ANNUNCIATOR
ALTITUDE PRESELECT DISPLAY
MACH DISPLAY
DECISION HEIGHT DISPLAY
VMO INDICATOR
VERTICAL TRACK ALERT ANNUNCIATOR
DECISION HEIGHT ANNUNCIATOR
ROLL ATTITUDE POINTER AND SCALE
IAS ACCEL TREND VECTOR
VERTICAL SPEED DISPLAY AND VECTOR
IAS/SCALE AND DISPLAY
GLIDESLOPE OR GLIDEPATH POINTER
IAS/MACH REFERENCE BUG
RADIO ALTITUDE DISPLAY
FMS SPEED BUG
LOCALIZER OR AZIMUTH POINTER AND RISING RUNWAY (NOTE 3)
VSPEED BUGS
VERTICAL SPEED TARGET
V SPEED REFERENCE
PITCH AND ROLL CMD CUE (NOTE 1)
AIRCRAFT SYMBOL
LOCALIZER OR AZIMUTH SCALE
INCLINOMETER
ATTITUDE SPHERE
API
M.372
283
LOC
GS
CAT2
ATT2
M
M
ASEL
23600
DH150
VTA
340
320
305
280
260
DH
20
20
10
10
10
10
6
2
1
1230
1
2
6
300
V=267
VS1220

NOTES:

1. THE CUE CAN BE CHANGED TO COMMAND BARS BY A PROGRAM PIN OPTION AT INITIAL INSTALLATION.
2. AT CAPTURE, THE MODE ANNUNCIATOR FLASHES FOR 5 SECONDS.
3. THE RISING RUNWAY IS ENABLED BY A PROGRAM PIN OPTION AT INITIAL INSTALLATION.

Figure 16A-20. ED-800 Displays and Annunciators (TCAS II Aircraft)

- Selected Course/Desired Track Pointer
- Selected Course/Desired Track Digital Display
- Course Deviation
- NAV Source Annunciator
- TO/FROM Pointer (FULL Mode)
- TO/FROM Annunciator (ARC, WX, MAP, or MAP/WX Modes)
- Preselected Course Pointer Preselected Course Annunciator
- Vertical Deviation Display
- Bearing 1/2
- Distance to VOR/DME/WPT
- WPT/DME Identifier and WPT Alert

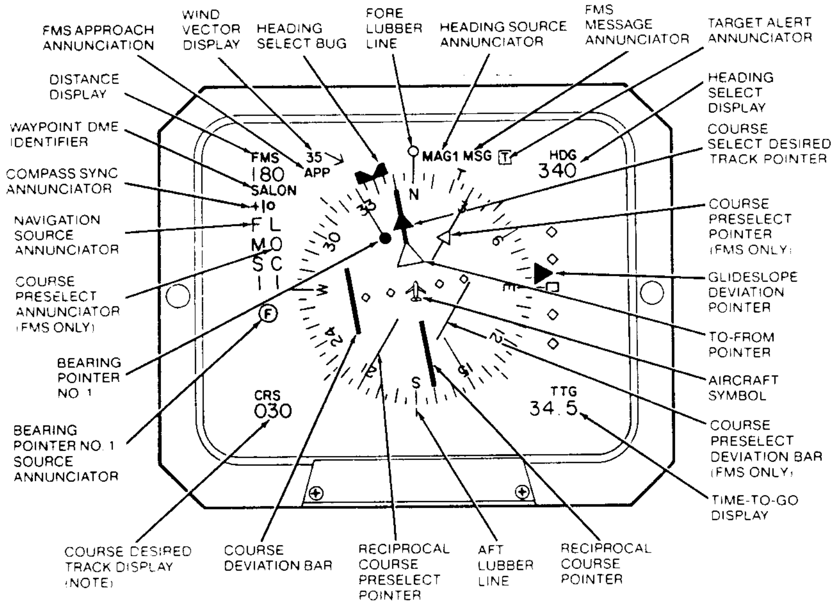
- NAV Data (TTG/GS/ET)
- Wind Vector Display
- BC Annunciation
- WX Target Annunciator
- Reversionary Source Annunciation
- Flags and Comparators
- FMS Alert Message
- FMS Approach Annunciator

WX, and MAP/WX Modes Only

- WX Range
- WX Mode Annunciator
- WX Return Display
- Lightning Detection
- Turbulance Mode

MAP and MAP/WX Modes Only

- Waypoints and Track Lines
- VOR/DME/VOR-DME Symbol Display
- Cross-Track Distance


FMS APPROACH ANNUNCIATION
WIND VECTOR DISPLAY
HEADING SELECT BUG
FORE LUBBER LINE
HEADING SOURCE ANNUNCIATOR
FMS MESSAGE ANNUNCIATOR
TARGET ALERT ANNUNCIATOR
DISTANCE DISPLAY
WAYPOINT DME IDENTIFIER
COMPASS SYNC ANNUNCIATOR
NAVIGATION SOURCE ANNUNCIATOR
COURSE PRESELECT ANNUNCIATOR (FMS ONLY)
BEARING POINTER NO 1
BEARING POINTER NO. 1 SOURCE ANNUNCIATOR
HEADING SELECT DISPLAY
COURSE SELECT DESIRED TRACK POINTER
COURSE PRESELECT POINTER (FMS ONLY)
GLIDESLOPE DEVIATION POINTER
TO-FROM POINTER
AIRCRAFT SYMBOL
COURSE PRESELECT DEVIATION BAR (FMS ONLY)
TIME-TO-GO DISPLAY
COURSE DESIRED TRACK DISPLAY (NOTE)
COURSE DEVIATION BAR
RECIPROCAL COURSE PRESELECT POINTER
AFT LUBBER LINE
RECIPROCAL COURSE POINTER
FMS
180
SALON
35
APP
MAG1 MSG
T
HDG
340
N
3
6
E
12
S
15
21
24
W
30
33
F L
M O
S C
F
CRS
030
TTG
34.5

NOTE WITH FMS SELECTED AND DURING COURSE PRESELECT THE CRS DISPLAY APPEARS DURING AND FOR 5 SECONDS AFTER ROTATION OF THE CRS KNOB AND THEN REVERTS BACK TO THE DTK DISPLAY

AD-15574 R2

Figure 16A-21. ED-800 EHSI Displays and Annunciators

COLOR CODING

The symbology is defined with the following logic of colors:

Amber

Reversionary information requiring special attention; miscompare annunciations.

Red

Flags, limits, data out of limits

White

Scales, compass rose/arc, armed modes, labels, reversion not affecting the safety; aircraft symbol on HSI; bearing pointer 2

Green

Present flight parameters; captured modes; NAV information from on-side; transfer annunciation on the side where the transfer is not made

Yellow

Aircraft symbol on ADI; NAV information from crossside

Cyan

Selected altitude and decision height; heading bug on HSI; selected [AS/MACH reference bug and digits; Wind/Elapsed time; bearing pointer 1; selected VS reference

Magenta

Preselected course arrow, digits and annunciator; associated TO waypoint data

Flashing information is used to alert the pilot.

INITIALIZATION AT POWER-UP

At power-up, the following initializations are provided:

NAV Source

On-side VOR selected

Selected Course

Synchro value

Selected Heading

Synchro value

Bearing

On-side

NAV Data

Ground speed

IAS/MACH Bug

IAS

Decision Height

Potentiometer value

ASEL

ASEL Box is empty until ASEL knob turned, then current altitude

Reversionary Switching

None

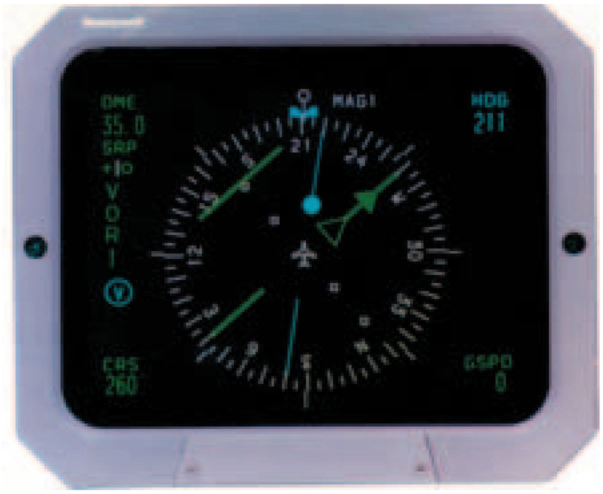


Figure 16A-22. EHSI Compass Display With VOR RMI Needle Overlay Selected

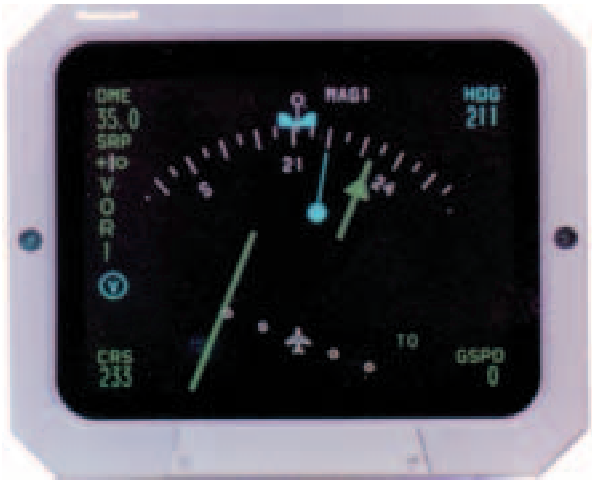


Figure 16A-23. EHSI Arc Mode Display With VOR RMI Needle Overlay Selected

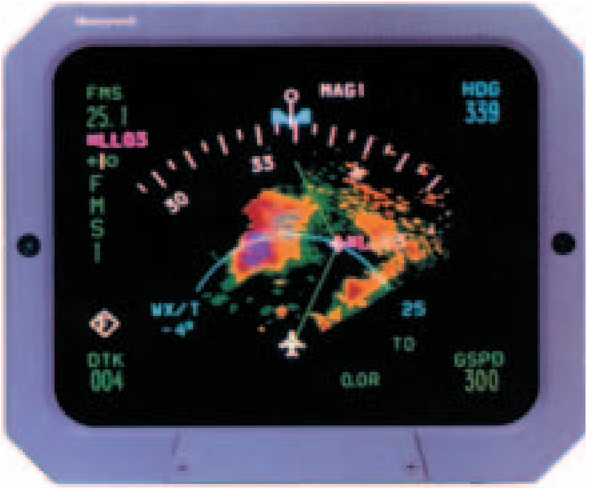


Figure 16A-24. EHSI MAP Mode With WX Radar

EFIS SELF-TEST

TEST Symbology on EADI

- The TEST display is invoked by the activation of the TEST pushbutton on the DC-810.
- The TEST pattern will be displayed only as long as the pushbutton is depressed.
- For the first 4 seconds, the TEST pattern shows all comparators.
- After 4 seconds, all comparators are removed and all flags are displayed with their corresponding logic.
- For the whole duration of the test, an amber TEST is displayed on EADI.
- If the aircraft is in the air, only radio altimeter is tested.
- If the aircraft is on the ground and the TEST function is performed, the SG will reset to power-up configuration after test.

TEST Symbology on EHSI

- The TEST display is invoked by the activation of the PUSH TEST pushbuffon on the DC-810.
- The TEST pattern will be displayed only as long as the pushbutton is depressed.
- For the first 4 seconds, the TEST pattern shows the IRS comparator.
- After 4 seconds, all flags are displayed with their corresponding logic.
- If the aircraft is in the air, only the radio altimeter will be tested.
- If the aircraft is on the ground and the TEST function is performed, the SG will reset to power-up configuration after TEST.

EFIS FAILURE MODES & REVERSIONS

There are several failure modes and reversions possible with the EFIS system.

Complete Symbol Generator Failure

Both tubes on the respective side go black (depending on the nature of the failure, there may be a red "SG" annunciated in the center of each tube)

Action: Either A or B below

a. Press the "SG reversion switchlight on the reversionary panel (Figure 16A-6). This will allow the opposide side SG to power its own side plus the failed side. All settings (heading, course, etc.) will have to be made on the operable S.G. sides

b. Select "SG" toward the failed side on the MC-800 MFD controller (Figure 16A-13). This will allow the MFD S.G. to power the failed side tubes. Settings (heading, course, etc.) can be made on the failed side. The MFD tube will now be blank.

Tube Failure

EADI Failure — The EADI will be blank. The EHSI will work normally.

Action: On the EDZ-817 display controller (Figure 16A-11), rotate the ADI dimming knob to the off position. This will bring up the ADI on the EHSI tube. The EHSI can now be displayed on the MFD by turning the MC 800 MFD controller knob to HSI

EHSI Failure — The EHSI will be blank. The EADI on the respective side will be working normally. The failed EHSI can be displayed on the MFD by turning the MC-810 controller knob to HSI (Figure 16A-13).

FMZ-800 FLIGHT MANAGEMENT SYSTEM

The Flight Management System consists of the control display unit (CDU), the navigation computer, various navigation sensors (GPS, IRS, VLF/OMEGA, VOR/DME) and an optional data loader. The FMS computer receives input from the various navigation sensors, processes it, prioritizes it, and then provides lateral and vertical navigation guidance for display and coupling to the DAFCS. Entry of V-speed bugs for display on the EFIS is also accomplished through the FMS CDU. Through use of its available navigation sensors and internal software, the FMS selects GPS/GPS as a primary source, then DME/DME, then VOR/DME and, lastly, AHRS/IRS. This position is then used to develop area navigation flight plans to any geographical point in the world. Routes are defined by the pilot from aircraft present position to a destination waypoint through a series of great circle legs connected by intermediate waypoints. With the addition of NZ-2000 database software, the FMS can be approved not only for enroute navigation, but for non precision approach procedures as well (VOR, NDB and if GPS sensors are installed, GPS approaches). The FMS also receives information from the ADC's, aircraft fuel system and AHRS. It uses this information to calculate various kinds of performance information (Figure 16A-26).

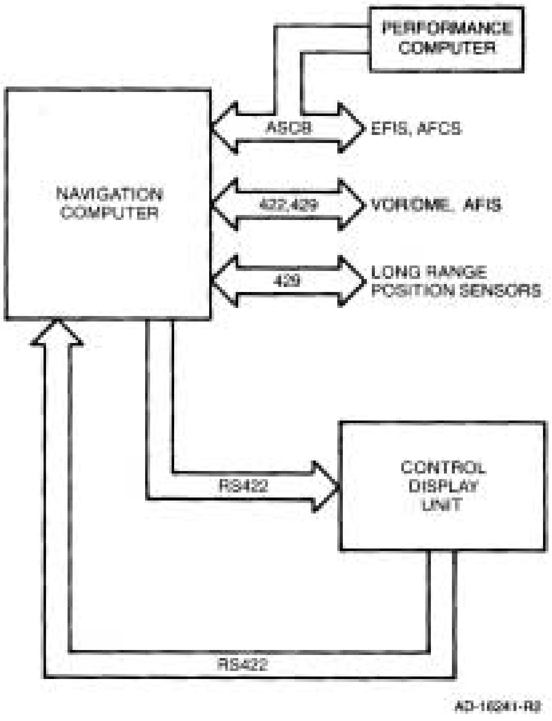


Figure 16A-25. FMS Block Diagram

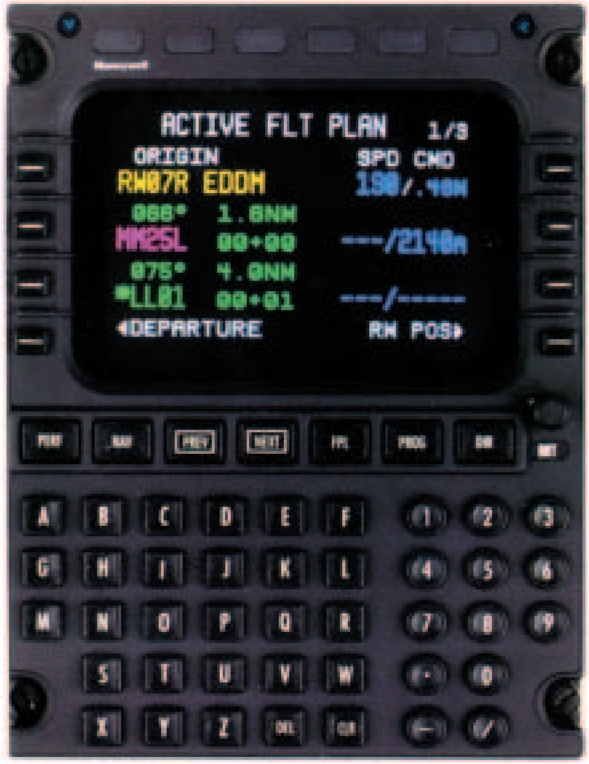


Figure 16A-26. FMS CDU

PRIMUS 870 WEATHER RADAR SYSTEM

Description and Operation

The Primus 870 system is an x-band, digital radar used for weather detection and ground mapping. Some installations have the optional LSZ-850 lighting sensor system.

The system uses one receiver/transmitter/antennae and two controllers.

Dual controllers allow both pilots to individually select radar mode, range, tilt and gain settings. On the right to left antenna scan, the system switches to what is selected on the left controller and updates the left HSI diaplay. On the reverse scan of the antenna, the right controller is selected and displayed on the right HSI. The radar controller displayed on the MFD is a function of the MFD controller (Figure 16A-27).

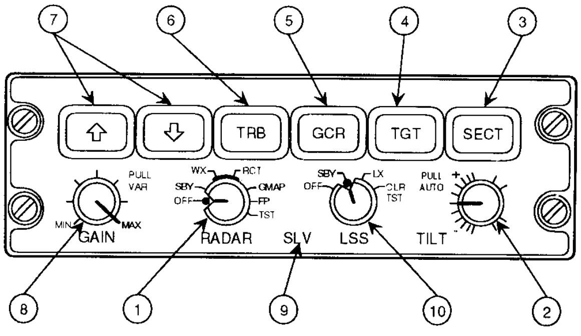


Figure 16A-27. Radar Controller

Radar

A rotary switch used to select one of the following functions:

OFF

Turns the radar system off. WX in amber is displayed in the mode field.

TBY (Standby)

The radar system is placed in standby, a ready state, with the antenna scan stopped, the transmitter inhibited and the display memory erased. STBY, in blue, is displayed in the mode field. If SBY is selected prior to the expiration of the initial R/T/A warmup period (approximately 45 seconds), the blue WAIT legend is displayed in the mode field. Upon completion of the WAIT period the system automatically switches to standby mode.

WX (Weather Radar)

Selecting WX places the radar system in the weather detection mode. The system is fully operational and all internal parameters are set for enroute weather detection.

If WX is selected prior to the expiration of the initial R/T/A warmup period (approximately 45 seconds), the blue WAIT legend is displayed. In WAIT mode, the transmitter and antenna scan are inhibited and the memory is erased. Upon completion of the warmup period, the system automatically switches to WX mode.The system, in preset gain, is calibrated as defined in Table 16A-2. Blue WX is displayed in mode field.

TABLE 16A-2. WEATHER DETECTION CALIBRATION SCHEME

Rainfall Rate (mm/hr)	Color
1-4 mm/hr	Green
4-12 mm/hr	Yellow
12-50 ~mm/hr	Red
Greater than 50 mm/hr	Magenta

RCT (REACT)

Adds the REACT feature to the WX display. REACT is an acronym for rain echo attenuation compensation technique and is a submode of the WX mode. Selecting RCT forces the system to preset gain. When REACT is selected, the blue RCT legend is displayed in the mode field. The REACT circuitry compensates for attenuation of the radar signal as it passes through rainfall. The cyan field indicates areas where further compensation is not possible. Any target detected within the cyan field cannot be calibrated and should be considered very dangerous. All targets in the cyan field are displayed as 4th level precipitation, magenta.

GMAP (Ground Mapping)

Selecting GMAP places the radar system in the ground mapping mode. The system is fully operational and all internal parameters are set to enhance returns from ground targets. RCT compensation is inactive.

CAUTION

> Weather-type targets are not calibrated when the radar is in the GMAP mode. Because of this, the pilot should not use the GMAP mode for weather detection

As a constant reminder that GMAP is selected, the blue GMAP legend is displayed and the color scheme is changed to cyan, yellow, magenta. Cyan represents the least reflective return, yellow is a moderate return and magenta is a strong return.

If GMAP is selected prior to the expiration of the initial R/T/A warmup period (approximately 45 seconds), the blue WAIT legend is displayed. In WAIT mode, the transmitter and antenna scan are inhibited and the memory is erased. Upon completion of the warmup period, the system automatically switches to GMAP mode.

FP (Flight Plan)

Flight plan places the WX transmitter in standby and allows the EHSI or MFD map range to be selected up to 1000 NM.

TST (Test)

Selects the radar test mode. A special test pattern is displayed to allow verification of system operation. The blue TEST legend is displayed in the mode field.

WARNING

> The tramsmitter is on and radiating X-band microwave energy in test mode Refer to maximum permissible exposure level (MPEL) in this Appendix.
>
> The system performs only functions of weather detection or ground mapping. It should not be used nor relied upon for proximity warning or anticolision proteciton.

FORCED STANDBY

Forced standby is an automatic, nonselectable radar mode. The controllers are wired to the weight-on-wheels (WOW) switch. The R/T/A is in forced standby mode when the aircraft is on the ground. In the forced standby mode, the transmitter and antenna scan are both inhibited, the memory is erased and the blue STBY legend is displayed in the mode field. When in the forced standby mode, pressing both range buttons simultaneously restores normal operation.

NOTE

> When dual weather radar controllers are installed, both controller's range buttons must be simultaneously pressed to restore normal operation.

Forced standby mode is a safety feature that inhibits the transmitter on the ground to eliminate the X-band microwave radiation hazard.

Tilt

A rotary coNtrol used to select tilt angle of antenna beam with relation to earth plane. Clockwise rotation tilts beam upward to +15 degrees; counterclockwise rotation tilts beam dowNWard to 15 degrees.

A digital readout of the antenna tilt angle is displayed on the EHSI.

PULL AUTO (Autotilt)

Pulling out on the TILT control knob causes the system to enter the autotilt mode. In autotilt the antenna tilt is automatically adjusted with regard to the selected range and barometric altitude. The antenna tilt automatically readjusts with changes in altitude and/or selected range. In autotilt, the tilt control can fine-tune the tilt setting by ±2 degrees.

Autotilt is annunciated by adding a letter A suffix to the tilt readout. The digital tilt readout always shows the actual tilt of the antenna regardless of the tilt command source (autotilt command or manual tilt command).

WARNING

> To avoid flying under or over storms, frequently select manual tilt to scan both above and below your flight level.
>
> Always use manual tilt for weather analysis.

Stabilization

This radar is normally attitude stabilized, and automatically compensates for roll and pitch maneuvers. Attitude stabilization may be deselected by pressing the TGT switch four times within 3 seconds. Stabilization is turned back on by again pressing TGT four times within 3 seconds.

Small antenna mounting misalignments or roll stabilization input offsets can be compensated for electrically. See radar pilot's manual.

SECT

Selects sector mode, allowing the radar display to be updated faster but with a 60-degree viewing angle.

NOTE

> In dual controller installation, selecting SECT on either controller faces SECT on both sides of the cockpit.

TGT

A button switch is used to enable and disable the radar target alert feature. Target alert is selectable in all but the 300-mile range. When selected, target alert monitors beyond the selected range and 7.5 degrees on each side of the aircraft heading. If a return with certain characteristics is detected in the monitored area, the target alert changes from the blue armed condition to the amber “T” warning condition. (Refer to the following target alert characteristics for a target description.) Thus, the pilot is advised of a potentially hazardous target directly in front of and outside the selected range. Upon receiving the amber warning, the pilot should select longer ranges to view the questionable target. Note that target alert is inactive within the selected range.

Selecting target alert forces the system into preset gain. Target alert can be selected in the weather radar (WX) and flight plan (FPLN) modes.

In order to activate target alert, the target must have the depth and range characteristics defined in Table 16A-3.

TABLE 16A-3. TARGET ALERT TARGET DEPTH AND RANGE CHARACTERISTICS

Selected Range (NM)	Target Depth (NM)	Minimum Target Range (NM)
10	2	10-50
25	4	25-75
50	4	50-100
100	6	100-150
200	6	200-250
300	Inactive	
Flight Plan (FPLN)	2	5-55

GCR

A button switch used to select the ground clutter reduction (GCR) mode of operation. The GCR mode is annunciated above the mode field with the GCR legend. The GCR mode can only be selected if the FUNCTION switch is in the WX position and the selected range is 50 miles or less. GCR is an advisory mode which reduces the display of ground clutter. Targets remaining on the display are very probably weather targets.

WARNING

Do not leave the radar in the GCR mode.

CAUTION

Even though the GCR circuits remove ground targets from the display, the ground is still present, it is simply not being displayed.

WARNING

GCR removes most of the ground targest from the display, but at the same time it removes some of the weather targets.

The GCR feature has the following limitations: It does not remove all of the ground and it does remove some of the weather. It is most effective dead ahead and its effectivity is reduced as the antenna scans away from dead ahead. The circuitry assumes reasonable tilt settings (looking down 15 degrees and turning on the GCR does not eliminate the ground return) for proper operation.

Selecting the 100-, 200- or 300-mile range or turbulence (TRB) detection mode turns off ground clutter reduction. The GCR legend is deleted from the mode annunciation and variable gain engages if previously selected. Subsequent selection of ranges of 50 miles or less reengages GCR. If not already selected, GCR forces the radar into preset gain.

TRB

A momentary-contact switch used to select the turbulence (TRB) detection mode of operation. The TRB mode can only be selected it the MODE switch is in the WX position and the selected range is 50 miles or less. The weather/turbulence mode is annunciated in the mode field with the blue WX/T legend. Areas of at least moderate turbulence are shown in soft white.

CAUTION

Turbulence can only be detected within areas of rainfall. The Primus® 870 weather radar does not detect clear air turbulence.

WARNING

Undetected turbulence may exist within any storm cell.

Selecting the 100-, 200- or 300-mile range turns off the turbulence detection. The "/T" is deleted from the mode annunciation and variable gain is engaged if previously selected. Subsequent selection of ranges of 50 miles or less re-engages turbulence detection.

Range

RANGE is a button control used to select one of six ranges (10, 25, 50, 100, 200 and 300 NM).

Gain

A single-turn rotary control and push/pull switch that is used to control the receiver gain. Pushing in on the GAIN switch causes the system to enter the preset, calibrated gain mode. Calibrated gain is the normal mode and is used for weather avoidance. In calibrated gain, the rotary portion of the GAIN control does nothing.

Pulling out on the GAIN switch causes the system to enter the variable gain mode. Variable gain is useful for additional weather analysis and for ground mapping. In WX mode, variable gain can crease receiver sensitivity over the calibrated level to show very weak targets or it can be reduced below the calibrated level to eliminate weak returns.

WARNING

Hazardous targets may be eliminated from the display with low settings of variable gain.

In GMAP mode, variable gain is used to reduce the level of the typically very strong returns from ground targets.

Minimum gain is with the control at its full counterclockwise (CCW) position. Gain increases as the control is rotated in a clockwise (CW) direction from full CCW to the 12:00 position. At the 12:00 position, both the gain and the sensitivity time control (STC) are at their maximum values. Additional CW rotation removes STC. At the full CW position, the gain is at maximum and the STC is at minimum.

NOTE

STC reduces the receiver gain at the start of the trace, and then increases it as the more distant returns are received. Thus, with STC a uniform display of cell strength is displayed for both near and distant cells.

The "V" legend annunciates variable gain. Selecting RCT, TGT, or GCR forces the system into preset gain. Preset gain is not annunciated.

SLV

A dead front annunciator that is only used in dual controller installations. With dual controllers, one controller can be slaved to the other by selecting OFF with the radar MODE switch. This slaved condition is annunciated with the SLV annunciator.

Note that both controllers must be off before the radar system turns off.

LSS (Option)

An optional four-position rotary switch which selects the separate LSZ-850 Lightning Sensor System operating modes.

Off

All power is removed from the lightning sensor system.

SBY (Standby)

Display of data from the lightning sensor system inhibited, but it is accumulating data.

LX (Lightning Sensor System)

LX is fully operational and displaying data on the indicator.

ANGLE-OF-ATTACK SYSTEM

General

A Safeflight angle-of-attack (AOA) system (Figure 16-28) is installed in the Hawker 800 XP. Refer to Chapter 15, "Flight Controls," Stall Warning and Identification, for additional information on the airflow angle sensors. It uses AOA information from the stall vanes.

Description

The Teledyne normalized AOA system senses local angle of attack along with airplane flap position, converts these inputs into lift information, and displays it on a normalized, ten-gradation, 0 to 1.0 dial. Zero lift is at 0 on the dial, and the maximum usable lift point, as defined by the *AFM* stall speed, is displayed as 1.0. In effect, the indicator presents lift as a percentage. With flap position information, the display is valid for all airplane configurations. Therefore, the system continuously displays the margin to the maximum usable lift point (1.0), regardless of airplane weight, bank angle, G loading, or airplane configuration.

The AOA indicator provides outputs to operate heads-up indexers only when the gear is down. The indexer displays the deviation in discrete steps by means of three lights. These lights are referenced to the V_{REF} speed target. An onspeed pointer indication (pointer centered on the index) produces a green doughnut in the indexer and an onspeed indication of the FAST/SLOW indicator in the EADI.

RADIO ALTIMETER

The EADI displays radio altitudes from 2,500 feet AGL to 0 feet. The display brightness is controlled remotely by a dimming potentiometer or a variable-dimming voltage independent of the integral instrument dimming control. The display is blank if the radio altitude is above 2,500 feet AGL or if the radio altitude valid input is invalid.

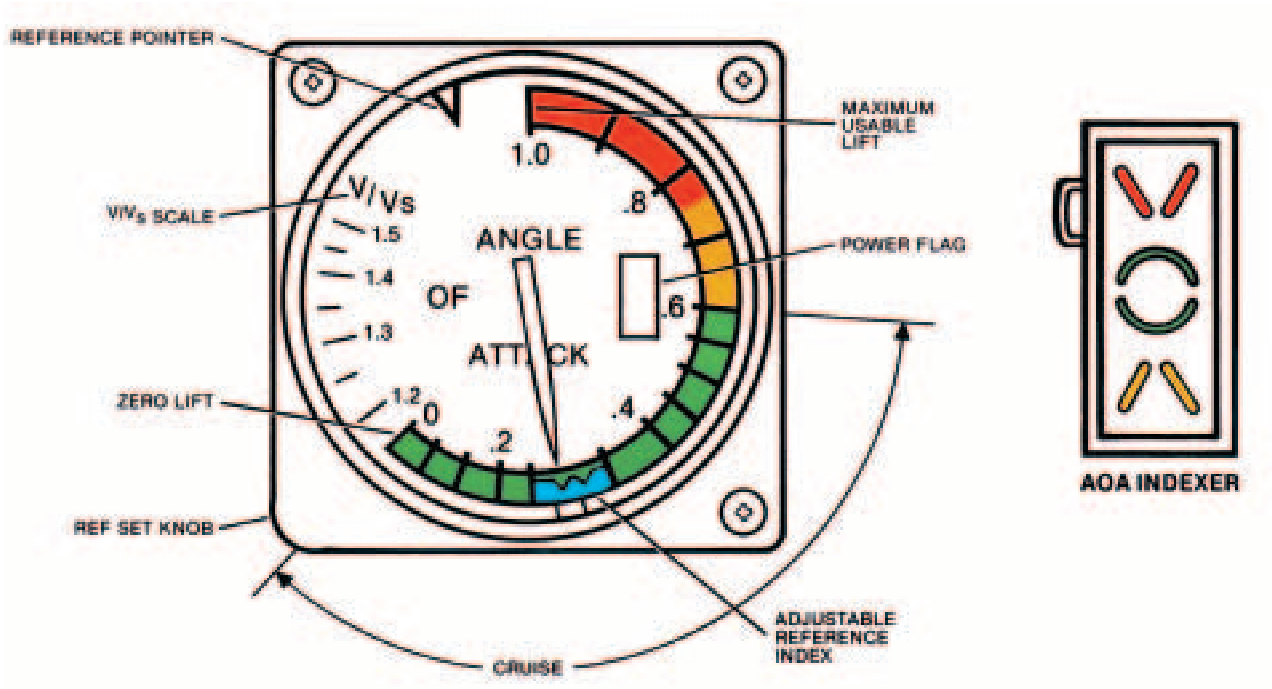


Figure 16A-28. Angle-of-Attack

STANDBY INSTRUMENTS

Standby Attitude Indicator

General

An electrically driven standby attitude indicator is located on the left instrument panel (Figure 16-29). In addition to pitch and roll information, glide-slope and localizer pointers and flags are provided which are driven by the No. 1 NAV receiver when a localizer frequency is tuned. The glide-slope and localizer pointers and flags are out of view when the No. 1 NAV receiver is not tuned to a localizer frequency. In the event of a power failure, a flag comes into view. A PULL TO CAGE knob is provided.

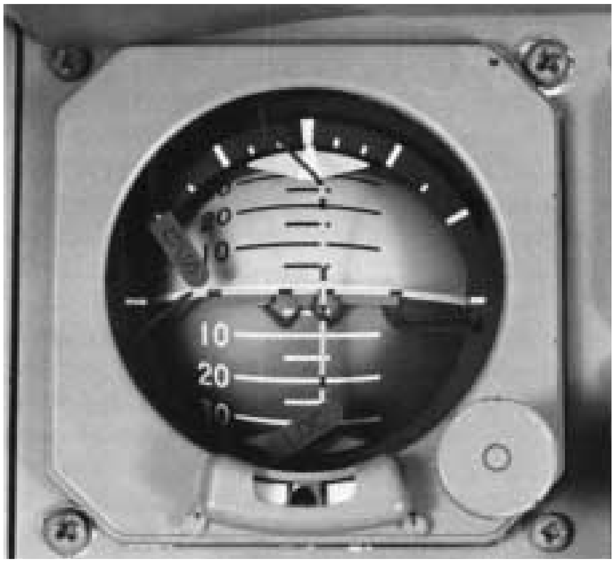


Figure 16A-29. Standby Attitude Indicator (Typical)

Operation

The standby horizon UNARMED/E BAT ON switchlight is located on the pilot's instrument panel. The top half of the switchlight capsule will illuminate UNARMED when power is first applied to the aircraft and the standby horizon will show red OFF flag.

Depressing the switchlight capsule to the armed E BAT ON position will extinguish the UNARMED annunciator and the standby horizon is powered from the aircraft's normal power system (28 VDC PS1 radio bus). The OFF flag will go out of view and the horizon will erect, provided that it is not caged.

NOTE

Leave the standby horizon caged for two minutes after ships power is on.

In the event of a power loss of the 28 VDC PS1 radio bus, the No. 3 battery will be automatically applied to the standby horizon to maintain the attitude display and internal lighting on the standby horizon. The E BAT ON annunciator on the bottom half of the standby horizon switchlight capsule will illuminate and an aural warning will sound. The aural warning may be canceled by depressing the horizon warn cancel pushbutton adjacent to the standby horizon.

In addition to the above, the standby altimeter and airspeed indicators will have their internal lighting provided by the No. 3 battery, and the No. 1 N_1 indicator will be powered by the No. 4 battery.

If normal power is restored to the aircraft, all systems will revert to the 28 VDC PS1 radio bus and the system will revert to an armed state.

The aural warning will also sound on the ground if power is removed from the aircraft and the standby horizon UNARMED/E BAT ON switchlight has been left in the ARMED position.

Standby Altimeter

A standby altimeter is provided on the center instrument panel. Altitude is displayed in the range of-1,000 to 50,000 feet by a single pointer read against a dial and a digital counter (Figure 16-30). The pointer makes one revolution of the dial per 1,000 feet. Gradations in 20-foot increments are numbered at 100-foot intervals. The counter shows ten-thousands and thousands of feet. A knob is provided to set the barascale counter, which simultaneously corrects the setting of the pointer and

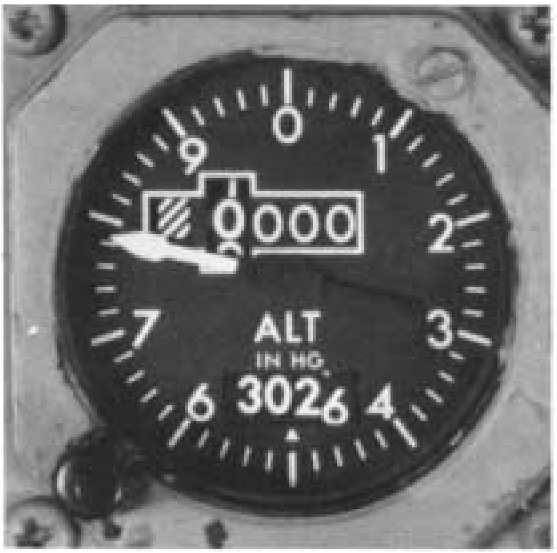


Figure 16A-30. Standby Altimeter

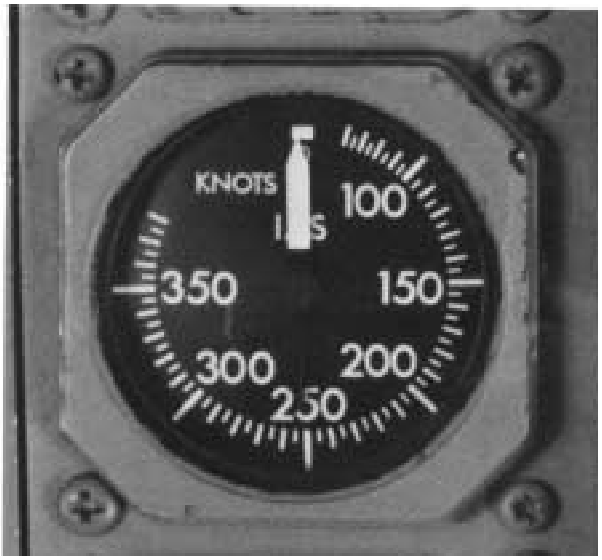


Figure 16A-31. Standby Airspeed Indicator

digital counter. The barascale counter readout may be in millibars or inches of mercury. In the event of power failure, the No. 3 battery powers the standby altimeter and airspeed internal lighting and the internal mechanical vibrator. If this vibrator is inoperative, the master MEL limitsaircraft operation to VMC conditions at arrival and departure airports.

Standby Airspeed Indicator

The standby airspeed indicator is located on the center instrument panel (Figure 16-31). It is connected to the right pitot system and the No. 8 static air vent. In the event of power failure, the No. 3 battery powers the standby airspeed indicator internal lighting.

POWER SOURCES

EFIS POWER SOURCES

CAPT—PE

F/O—PS 2

MFD—PS 1

AHRS

#1 AHRS—P.E. with a secondary feed also from P.E.

If P.E. is lost, #4 battery will power #1 AHRS for 2.5 minutes.

NOTE

> #1 compass card also requires XE bus power

#2 AHRS —PS-2 with a secondary feed from PS-1. if PS-1 and PS-2 are lost, then P.E. will power #2 AHRS for 2.5 minutes.

NOTE

#2 compass card also requires XS-1 bus power

Air Data

ADC #—P.E.

NOTE

The pilots altimeter also requires XE bus power

ADC #2—PS-2

NOTE

The copilots altimeter also requires XS-1 bus power

DFZ-800 FLIGHT CONTROL SYSTEM

LIMITATIONS

1. Maximum speed limit for ~autopilot operation is unchanged from the airplane maximum airspeed limit (V_{MO}/M_{MO}).
2. Do not use autopilot under 100 feet above the terrain. DH must not be less than 200 feet with special authorization.
3. Do not use autopilot or yaw damper during takeoff or landing.
4. Takeoff with one (1) AHRS inoperative or in Basic Mode is permitted only if a third Attitude and Heading Reference source is available for Attitude and Heading display in place of failed or degraded AHRS.
5. A/P—Y/D engagement prohibited if dispatch with only one AHRS or DADC operational. (Autopilot and Yaw Damper may be used if one AHRS or DADC fails inflight).
6. Do not use the AHRS for navigation (heading or attitude) after a complete power loss of more than 2-3 minutes in flight.
7. Do not engage yaw damper prior to reaching 400 feet above terrain after take-off.
8. Auto pilot must be disengaged for go-around operation.

CHAPTER 17
MISCELLANEOUS SYSTEMS

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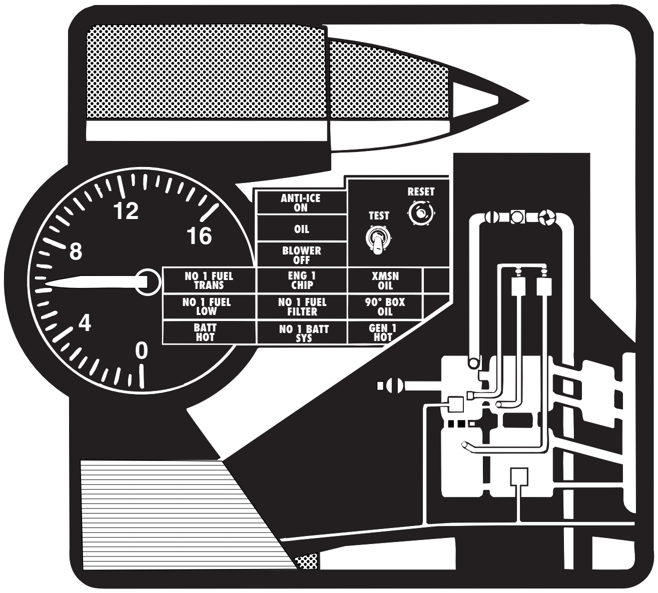
CHAPTER 17
MISCELLANEOUS SYSTEMS



INTRODUCTION

The Hawker 800 XP oxygen system provides oxygen for crew and passenger use from two high-pressure steel cylinders. In flight, oxygen is available to the crew at all times. The passengers' oxygen is available automatically at high cabin altitudes or manually anytime at the pilot's discretion. Oxygen for therapeutic purposes is available at all times at selected cabin outlets. A portable oxygen system (smoke set) is stowed in the cockpit.

OXYGEN SYSTEM

GENERAL

The oxygen system provides an oxygen supply to both pilots with a provision for a third crewmember. Provision is also made for fitting single or dual dropout passenger masks in the vestibule, the toilet compartment, and the passenger cabin. Therapeutic outlets are installed on the airplane as standard equipment.

The system includes two oxygen cylinders (a third is optional), a pressure regulator, and crew and passenger oxygen masks. Pressure-demand regulators are incorporated into the crew masks. The passenger masks are constant-flow. In addition, the system incorporates a master supply valve, the barometric valve, and a passenger supply valve (Figure 17-1).

Table 17-1. AVERAGE TIME OF USEFUL CONSCIOUSNESS

15,000 to 18,000 feet	30 minutes or more
22,000 feet	5 to 10 minutes
25,000 feet	3 to 5 minutes
28,000 feet	2 1/2 to 3 minutes
30,000 feet	1 to 2 minutes
35,000 feet	30 to 60 seconds
40,000 feet	15 to 20 seconds
45,000 feet	9 to 15 seconds

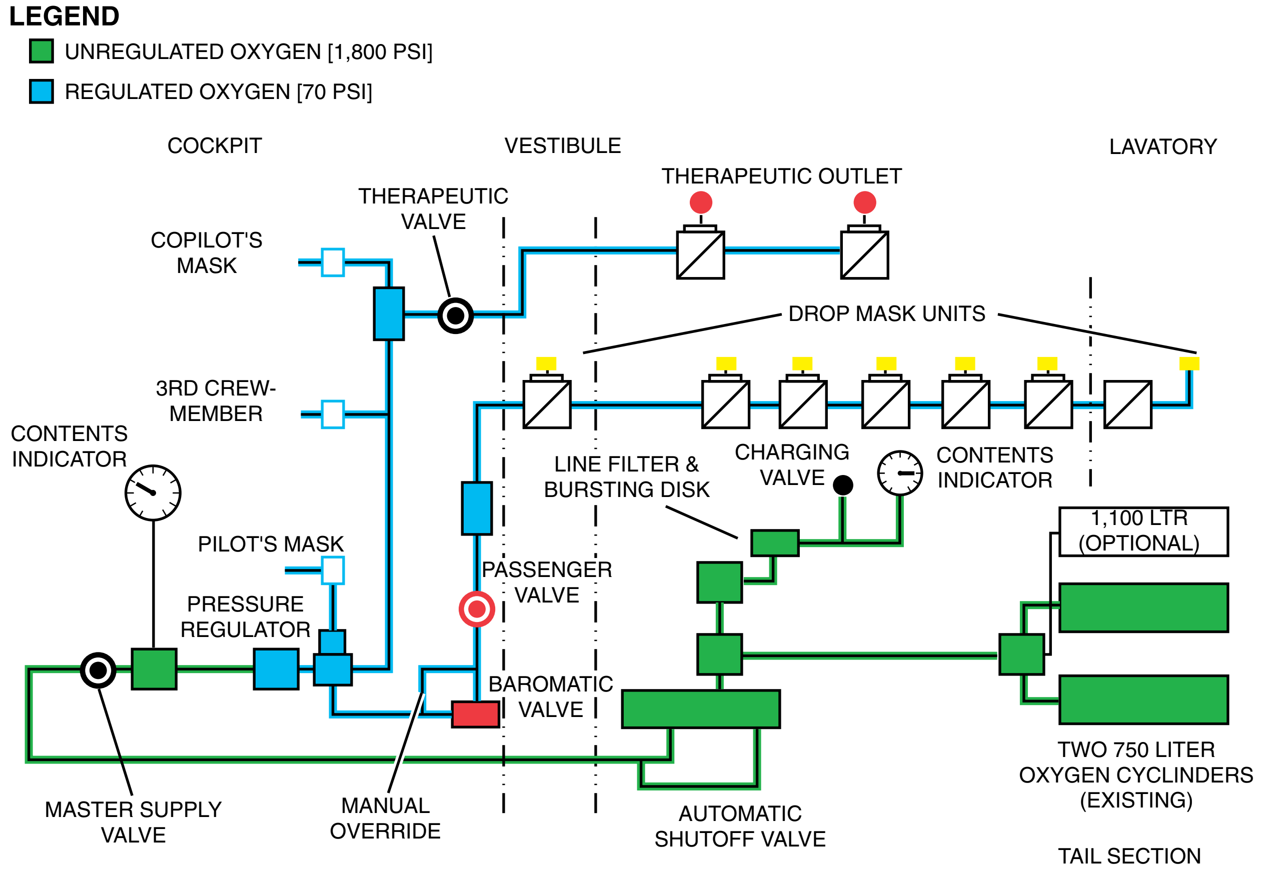


Figure 17-1. Oxygen System

Table 17-1 depicts the average time of useful consciousness (time from onset of hypoxia until loss of effective performance) at various altitudes.

OXYGEN CYLINDER ASSEMBLY

Two 750-liter oxygen cylinders store oxygen at 1,800 psi at 70° F. They are located in the rear equipment bay (Figure 17-2). A third 1,100-liter cylinder may be installed when increased system capacity is required. Each cylinder has a shutoff valve which is normally lockwired to the open position.

All cylinders are filled through one charging valve. A contents indicator (gage) (Figure 17-3) and a line filter, which incorporates a bursting disc, are parts int he charging supply line; therefore, cylinder pressure is present at the gage and the filter. If excess pressure is applied, the disc bursts and the cylinders vent to the atmosphere. Additionally, the oxygen cylinders incorporate an automatic shutoff valve installed in the supply line. The shutoff valve automatically closes should there be a break of the supply line downstream of the valve.

OXYGEN SYSTEM CONTROLS

The oxygen panel is mounted on the base of the left console (Figure 17-4). The panel has a master SUPPLY valve, a PASSENGER SUPPLY EMERGENCY valve control knob which controls the manual operation of a barometric valve, an OXYGEN CONTENTS gage, and two TEST connections for the barometric valve. The master SUPPLY valve controls the oxygen supply from the cylinders. Turning the valve to the ON position provides oxygen to the pressure regulator. The regulator reduces the 1,800-psi oxygen down to 70 psi. If the pressure exceeds 90 psi, a relief valve in the regulator bleeds the excess into the cabin. Also, with the SUPPLY valve in the ON position, the OXYGEN CONTENTS gage reads direct system high pressure, thus allowing the pilot to monitor the oxygen system quantity. Gage markings are FULL, 3/4, 1/2, 1/4, and EMPTY.

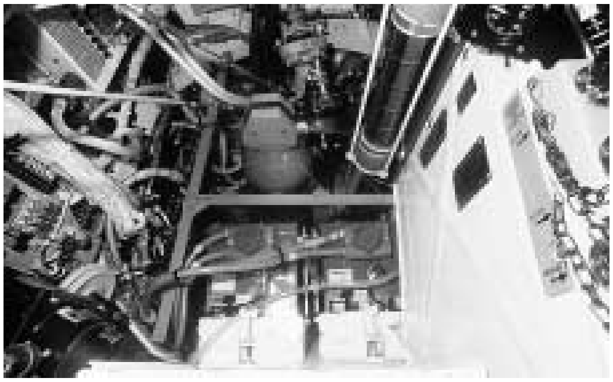


Figure 17-2. Oxygen Cylinders

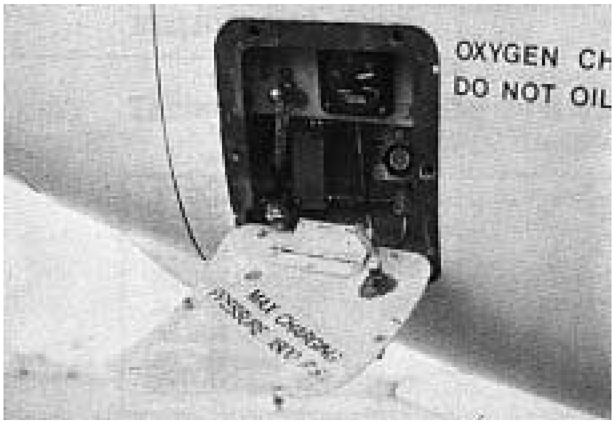


Figure 17-3. Oxygen Servicing Panel

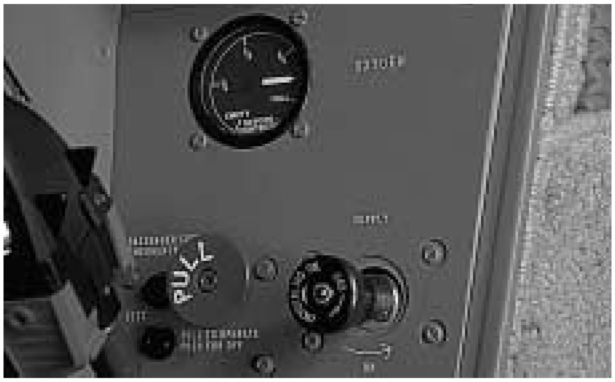


Figure 17-4. Oxygen System Controls

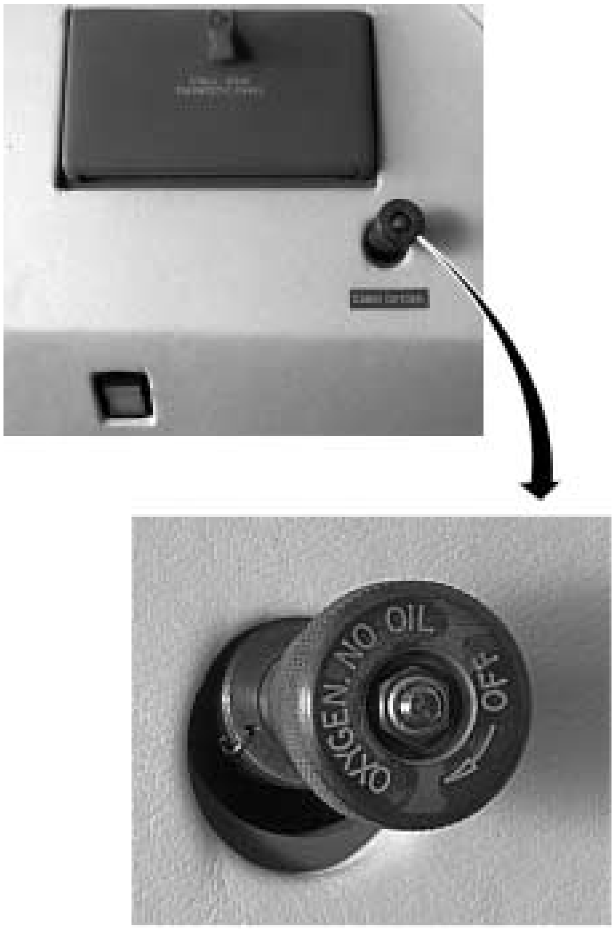


Figure 17-5. Passenger Supply Valve

With the master SUPPLY valve open, oxygen is available to the crew's pressure regulator and to the therapeutic outlets. A barometric valve located under the oxygen panel automatically controls delivery of oxygen to the passenger masks via a passenger supply (on/off) valve (Figure 17-5) which is on the forward face of the left bulkhead in the cockpit.

CREW OXYGEN SYSTEM

Two EROS masks/regulators are provided. Each mask has a pneumatic head harness and a regulator (Figure 17-6). The regulator has a two-position N–100% PUSH rocker selector switch and a PRESS TO TEST/(turn for) EMERGENCY knob. For the mask to operate, it must be plugged into the quick-release socket.

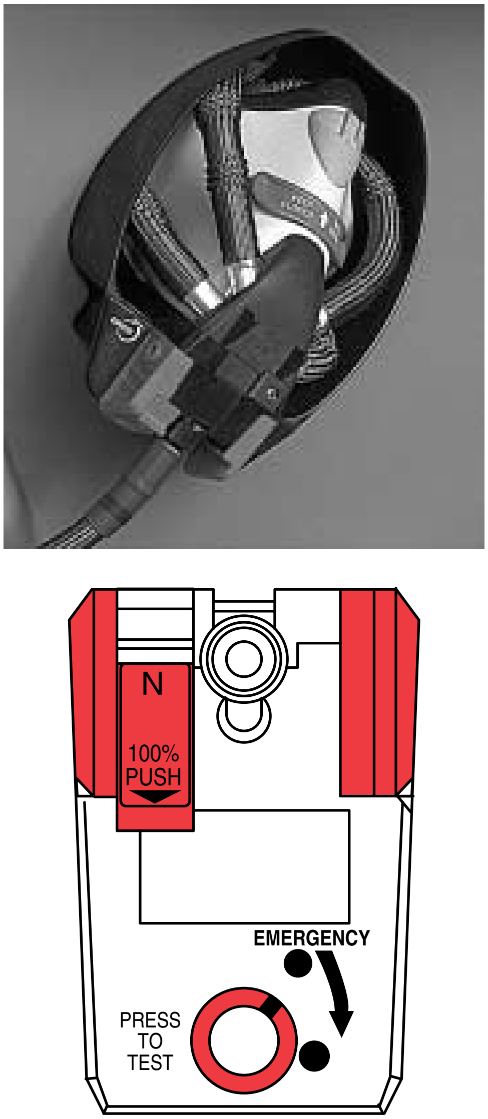


Figure 17-6. EROS Oxygen Mask

The mask is removed from storage by grasping and depressing the red harness control inflation plates of the regulator. This opens a valve and allows oxygen to flow from the supply to inflate the harness (see Figure 17-6). As the inflated harness is fitted over the head, releasing the control plates will allow the oxygen from the harness to vent. As the harness deflates, the mask is held firmly against the user's face.

With the selector switch positioned to N (normal), a mixture of air and oxygen is available up to a cabin altitude of 30,000 feet. Above this altitude, 100% oxygen is available and, at altitudes above 35,000 feet, the supply is at positive pressure. By selecting 100%, pure oxygen is provided at all altitudes. Also, by either turning the regulator knob to EMERGENCY or by depressing the knob to the PRESS TO TEST position, pure oxygen is supplied at a positive pressure.

An in-line flow indicator indicates DEBIT when oxygen flows.

PASSENGER OXYGEN SYSTEM

With the master SUPPLY valve open, oxygen passes to the barometric valve. This valve opens automatically when the cabin altitude reaches 12,000 ±500 feet, allowing oxygen to flow to each of the passenger masks via a normally open passenger supply shutoff valve. Supply pressure operates the dropout (deployment) mechanism, which opens the mask compartment door, allowing the passenger mask to drop out (Figure 17-1). When the mask is pulled down for use, a retaining clip releases a valve, and oxygen flows at a continuous rate to the wearer. Flow to an individual mask can be stopped by reinserting the retaining clip.

During automatic operation of the passenger oxygen system, the barometric valve moves the PASSENGER SUPPLY EMERGENCY valve (red) control knob upward to its on position (see Figure 17-4). If necessary, the pilot can manually supply oxygen to the system at any cabin altitude by pulling the control knob to the on (PULL TO OPERATE) position. The oxygen supply may be terminated by pushing the knob to the off (PUSH FOR OFF) position whenever the cabin altitude falls below 11,000 feet.

THERAPEUTIC OXYGEN OUTLETS

The therapeutic oxygen outlets are located in the cabin (see Figure 17-7). The outlets are self-sealing and receive their oxygen supply via the master SUPPLY valve. The masks used

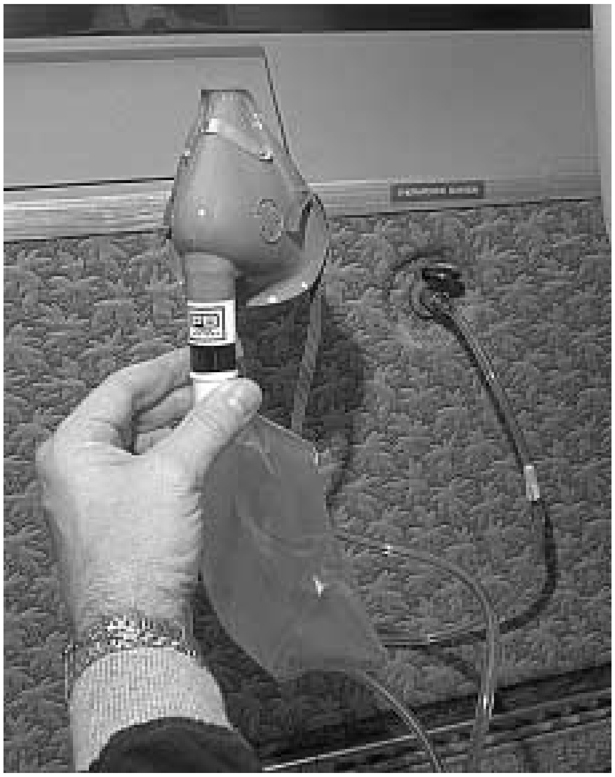


Figure 17-7. Therapeutic Oxygen Outlet and Mask

with the therapeutic oxygen outlets are normally stowed in the vestibule cabinet. When a mask is connected to an outlet, oxygen flow is shown by an indicator in the mask hose.

A therapeutic oxygen system shutoff valve is provided as an option on the bulkhead behind the captain's seat.

PORTABLE OXYGEN SMOKE SET

The portable oxygen smoke set consists of one oxygen cylinder and a mask normally stowed on the cockpit bulkhead behind the copilot (Figure 17-8). The cylinder should be pressurized to 1,800 psi which supplies oxygen for approximately 15 minutes. The cylinder incorporates an ON–OFF control knob and shutoff valve, a pressure indicator, a charging connection, a high-pressure relief valve, and two plug-in outlets with dust covers. To use the smoke set, the crewmember must turn the control knob to the fully ON position, remove the dust cover from an outlet, and don the mask.

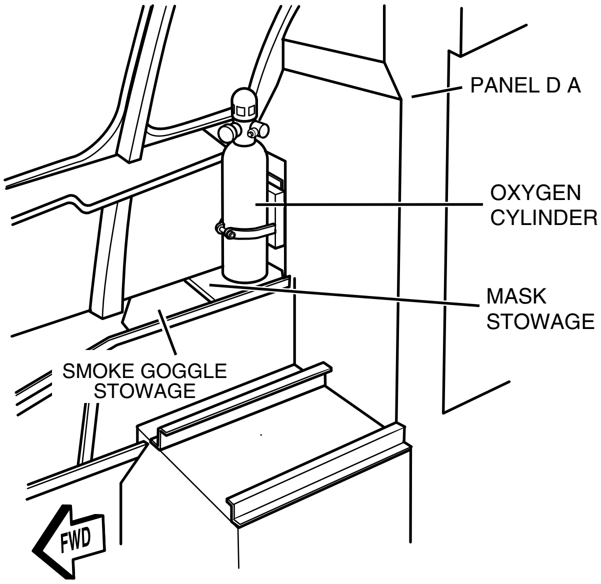
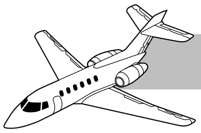


Figure 17-8. Portable Oxygen Smoke Set



QUESTIONS

1. The fully serviced oxygen cylinder pressure at 70° F is:

A. 70 psi
B. 90 psi
C. 1,800 psi
D. 2,000 psi

2. The charging OXYGEN CONTENTS gage reads:

A. Direct system low pressure
B. Electrically derived system low pressure
C. Direct system high pressure
D. Electrically derived system high pressure

3. The crew OXYGEN CONTENTS gage reads:

A. Direct system low pressure
B. Electrically derived system low pressure
C. Direct system high pressure
D. Electrically derived system high pressure

4. The control the pilot uses to manually supply oxygen to the passenger oxygen system is the:

A. Master SUPPLY valve control knob
B. PASSENGER SUPPLY EMERGENCY valve (red control knob)
C. PULL TO OPERATE switch
D. Passenger supply on–off valve

5. The valve that controls the oxygen supply to the therapeutic outlets is the:

A. Master SUPPLY valve
B. Passenger supply on–off valve
C. Barometric valve
D. PASSENGER SUPPLY EMERGENCY valve

6. The passengers' constant-flow masks deliver oxygen:

A. At all times when the system is activated and the mask lanyard is pulled
B. Upon inhalation only when the system is activated and the mask lanyard is pulled
C. When the system is activated and the mask falls from the stowage container
D. When the passenger supply on–off valve is positioned to ON

7. The passenger oxygen masks are automatically deployed and oxygen is available to the passengers when the:

A. Master SUPPLY valve is positioned to ON
B. Passenger supply on–off valve is positioned to ON
C. Cabin altitude reaches 12,000 ±500 feet and the master SUPPLY and passenger supply ON–OFF valves are in the ON positions
D. Cabin altitude reaches 12,000 ±500 feet, the master SUPPLY valve is ON, and the passenger supply on–off valve is positioned to either on or off

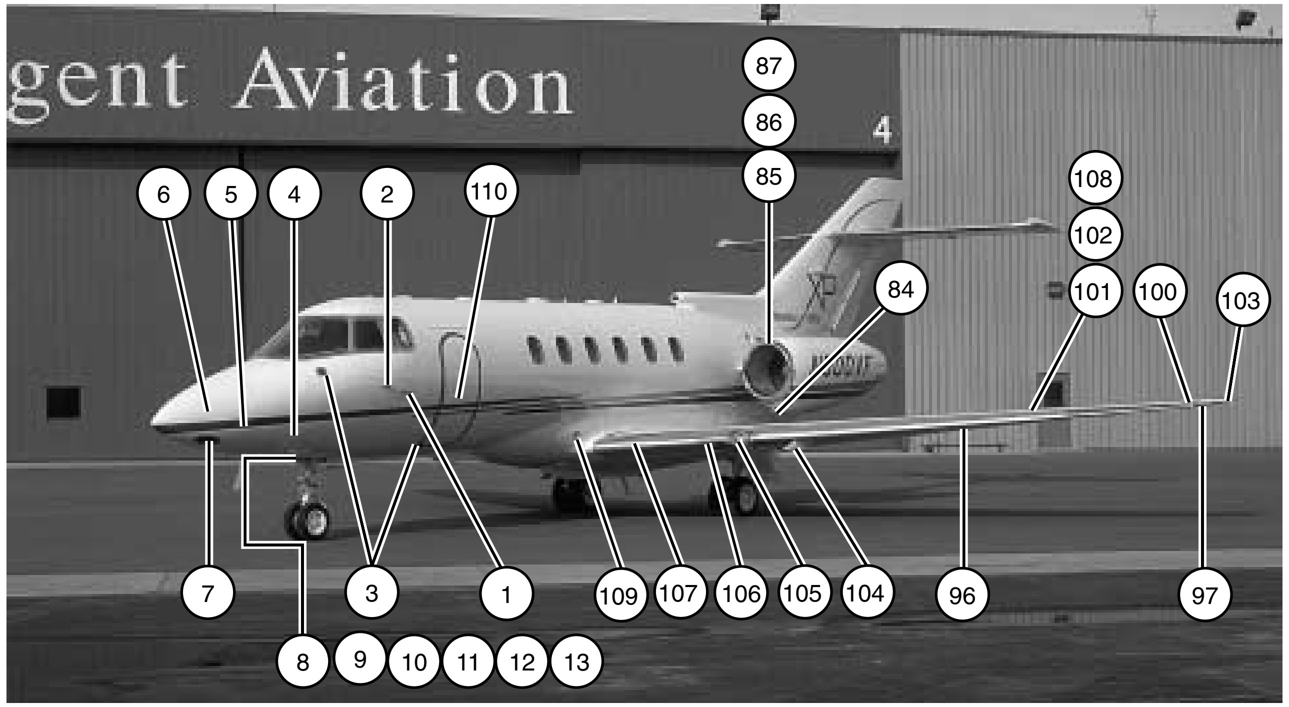
8. The portable oxygen smoke set supplies oxygen to the wearer for approximately:

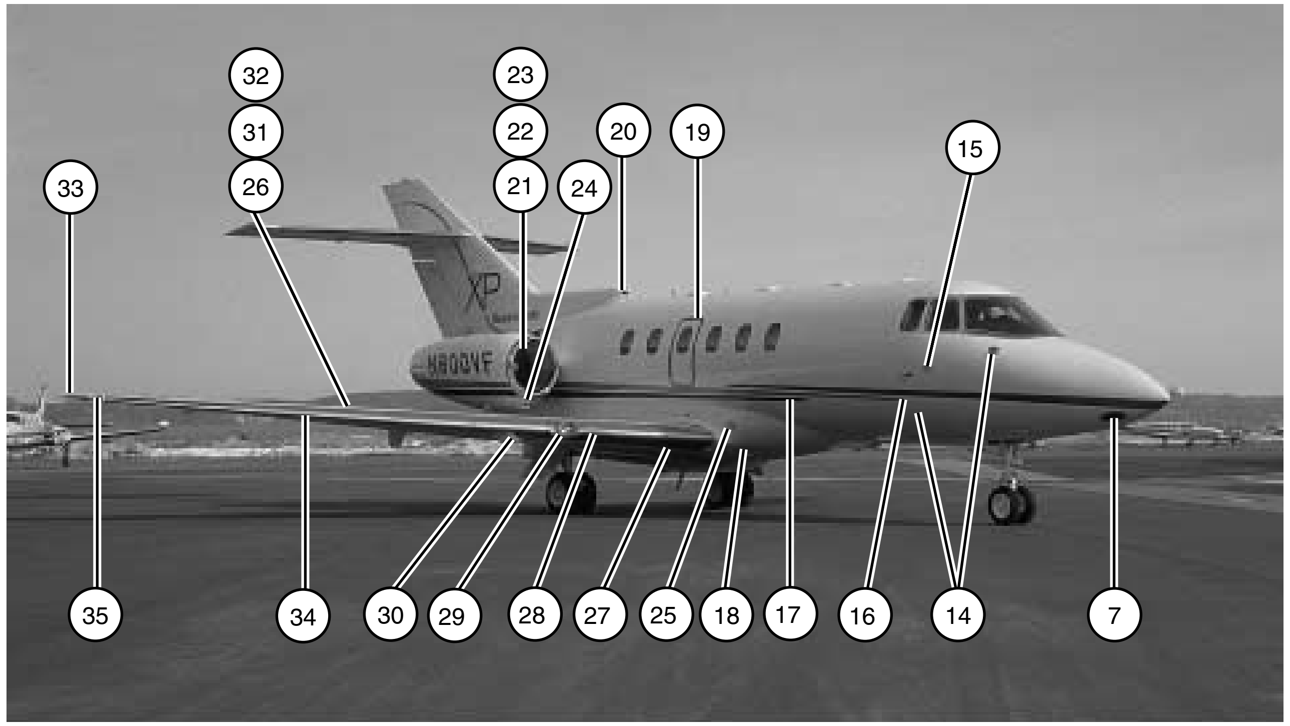
A. 5 minutes
B. 10 minutes
C. 15 minutes
D. 20 minutes

WALKAROUND

The following section is a pictorial walkaround. It shows each item called out in the exterior power-off preflight inspection. The fold-out pages at the beginning and end of the walkaround section should be unfolded before starting to read.

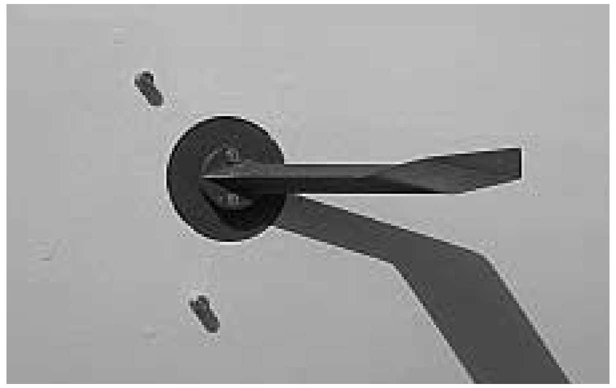
The general location photographs do not specify every checklist item. However, each item is portrayed on the large-scale photographs that follow.



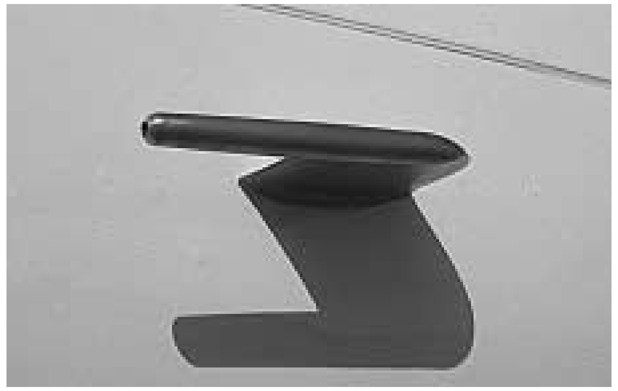


WALKAROUND

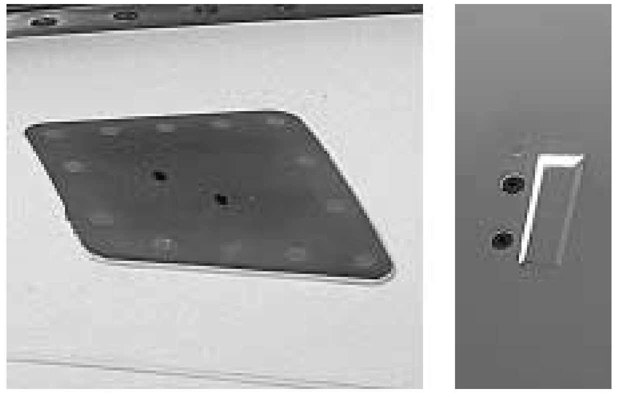
FUSELAGE NOSE—LEFT SIDE



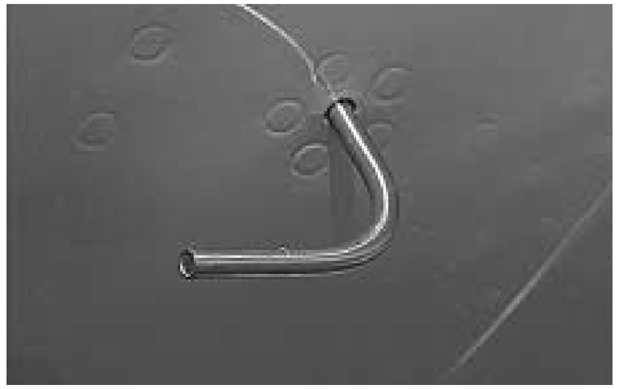
1. **AIRFLOW ANGLE SENSOR VANE**—CHECK FREEDOM OF MOVEMENT



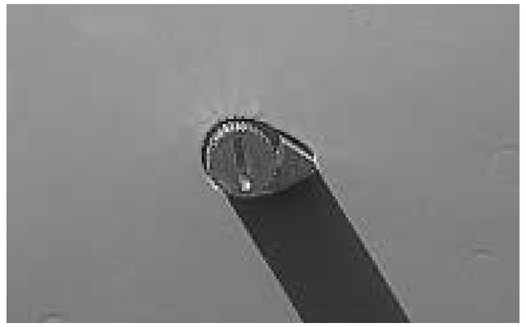
2. **PITOT PROBE**—COVER REMOVED, HOLE CLEAR



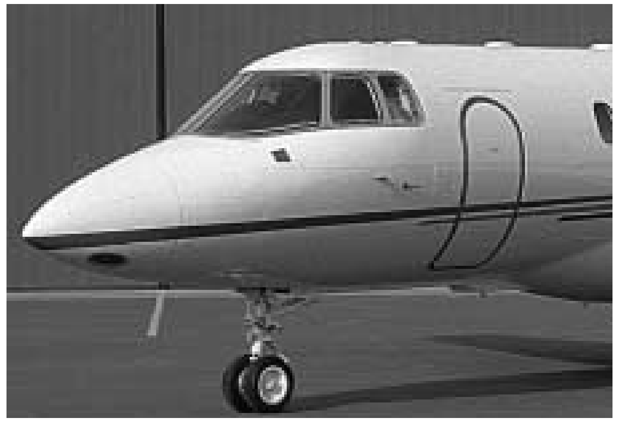
3. **STATIC VENTS**—PLUGS REMOVED, HOLE CLEAR (2 PLATES)



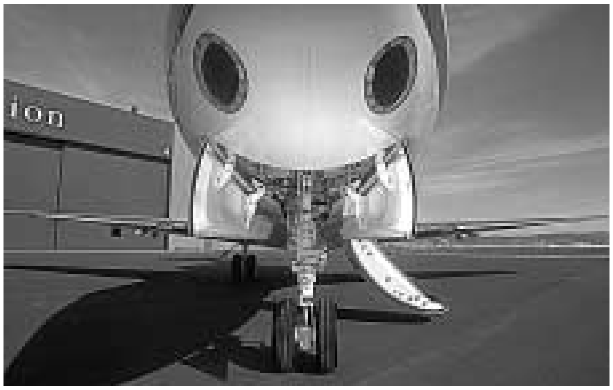
4. **AIR TEMPERATURE PROBE**—SECURE, UNDAMAGED, CLEAR



5. **ROTARY ICE DETECTOR**—FREEDOM OF ROTOR

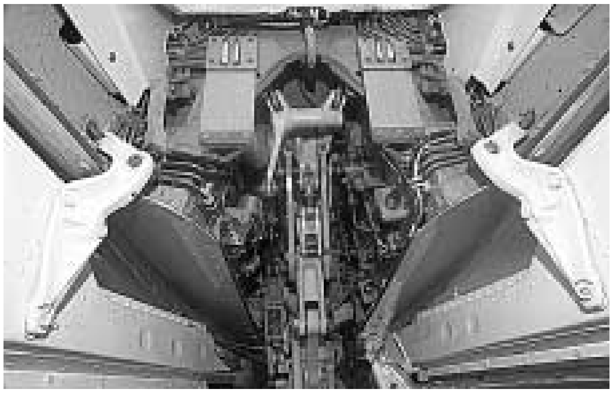


6. **RADOME**—UNDAMAGED, CLEAN, LOCKED

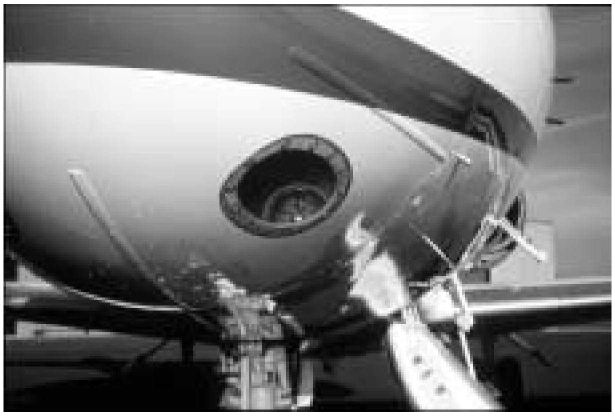


7. **NOSE TAXI LIGHTS**—LENSES CLEAN, UNDAMAGED, SECURE

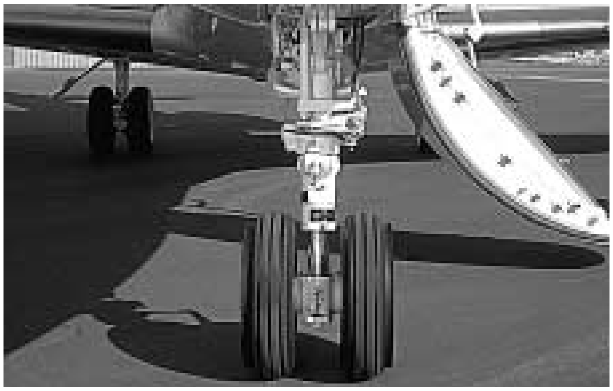
NOSE GEAR



8. **NOSE WHEEL WELL**—AUXILIARY HYDRAULIC FLUID CONTENTS, NO EVIDENCE OF LEAKS, GENERAL CONDITION
9. **GROUND LOCK**—REMOVED

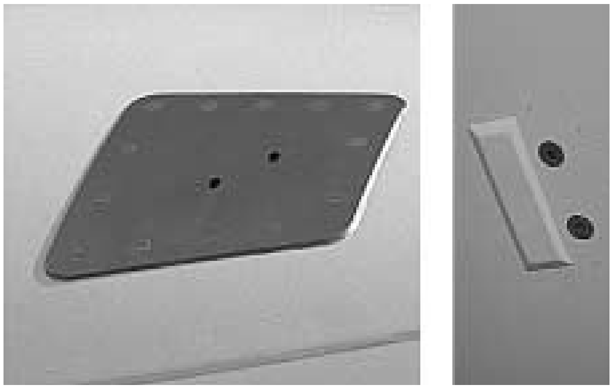


10. **NOSE WHEEL WELL DOORS**—CLOSED AND MANUAL RELEASE SECURE

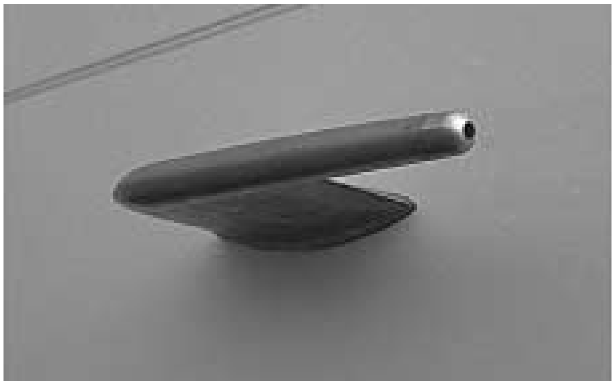


11. **NOSE STEERING PIN**—INSTALLED
12. **NOSE GEAR STRUT**—CLEAN, NO FLUID LEAKS
13. **TIRES**—GENERAL CONDITION

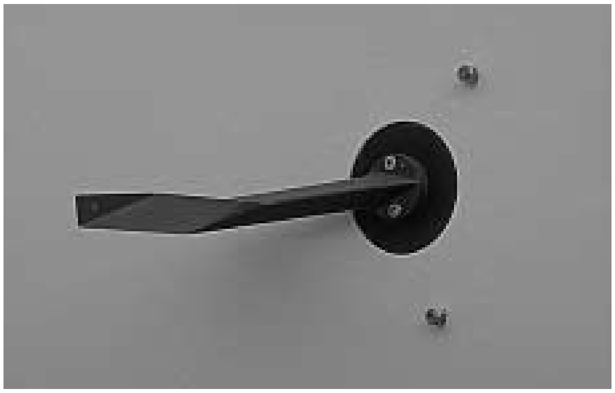
FORWARD FUSELAGE—RIGHT SIDE



14. **STATIC VENTS**—PLUGS REMOVED, HOLES CLEAR (2 PLATES)



15. **PITOT PROBE**—COVER REMOVED, HOLE CLEAR



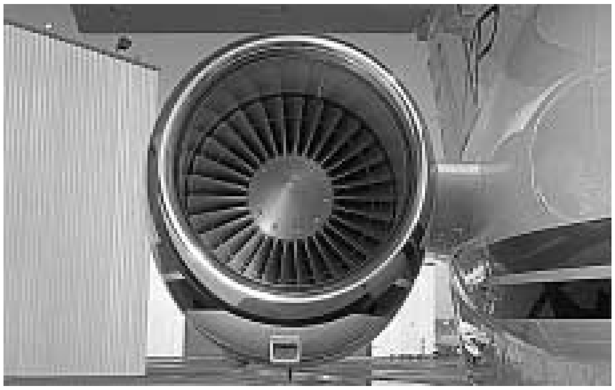
16. **AIRFLOW ANGLE SENSOR VANE**—CHECK FREEDOM OF MOVEMENT

17. **VENTURI OUTLET**—CLEAR

18. **SAT/TAS PROBE**—CHECK

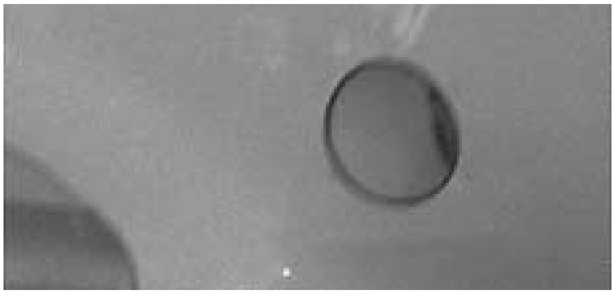
19. **EMERGENCY EXIT**—CLOSED
20. **DORSAL AIR INTAKE**—COVER REMOVED

RIGHT ENGINE

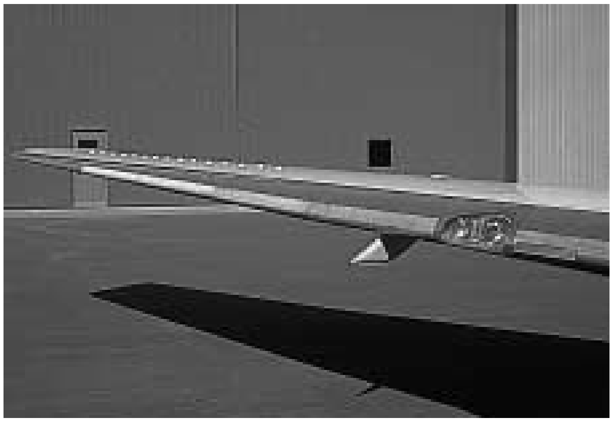


21. **INTAKE**-COVER REMOVED, NO FOREIGN OBJECT DAMAGE
22. **P_2T_2 SENSOR**—CLEAN, UNDAMAGED, SECURE
23. **FAN BLADES**—UNDAMAGED
24. **STARTER**—GENERATOR COOLING AIR INTAKE-UNOBSTRUCTED

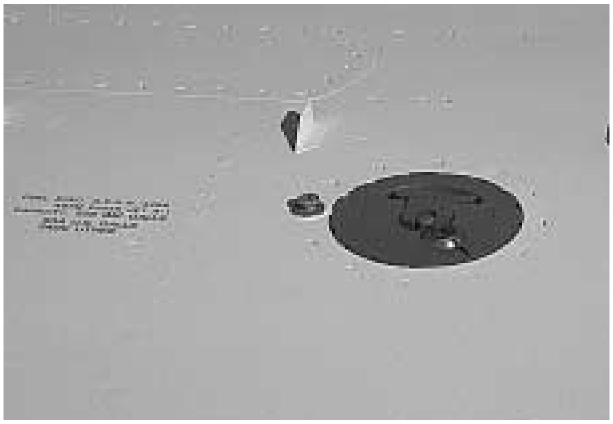
RIGHT WING



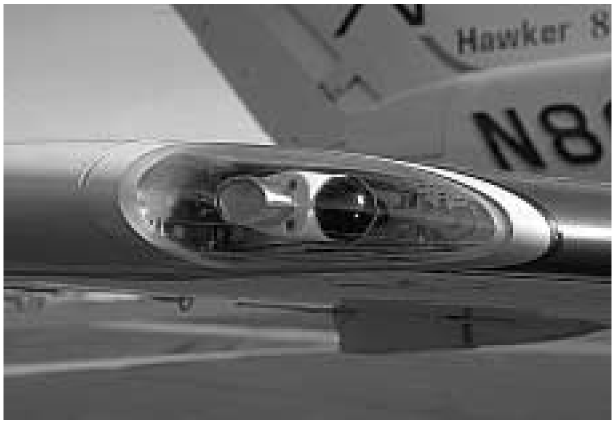
25. **WING ICE INSPECTION LIGHT**—CHECKED



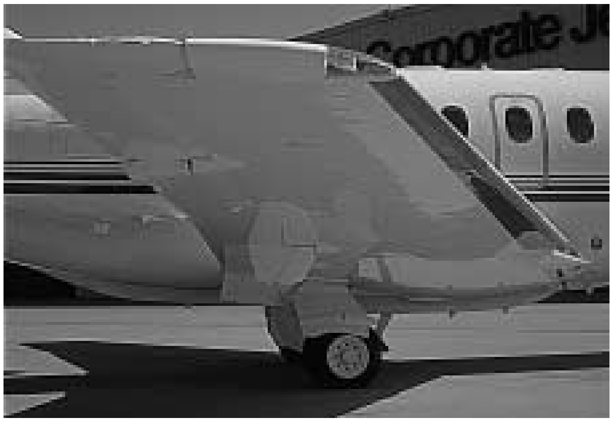
26. **TOP SURFACE**—GENERAL CONDITION
27. **TKS PANELS**—CLEAN, UNDAMAGED
28. **STALL STRIP**—SECURE, UNDAMAGED
29. **LANDING AND TAXI LIGHTS**—CONDITION OF LIGHTS, COVER CLEAN, UNDAMAGED
30. **VORTILON**—SECURE, UNDAMAGED



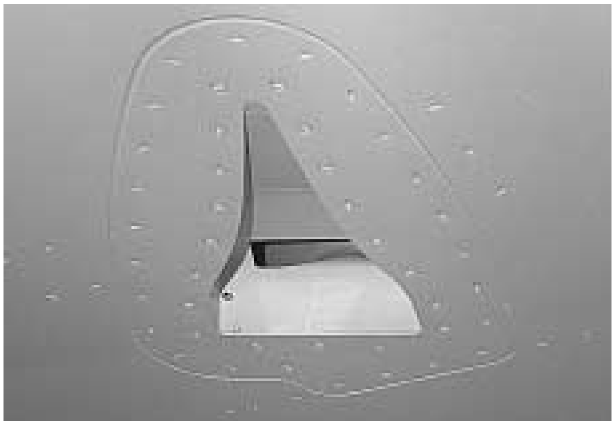
31. **FUEL CAP**—SECURE
32. **VORTEX GENERATORS**—SECURE



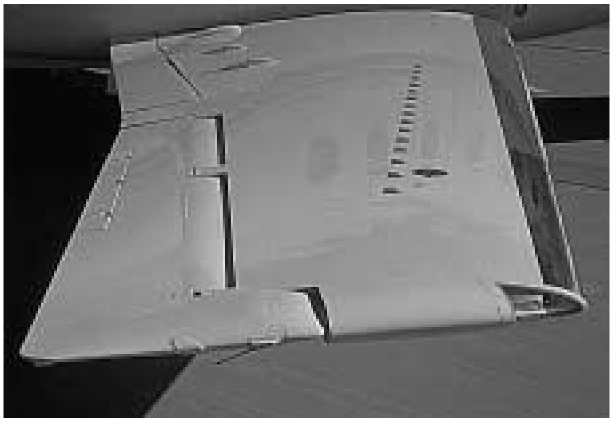
33. **NAVIGATION AND STROBE LIGHTS**—COVER CLEAN, UNDAMAGED



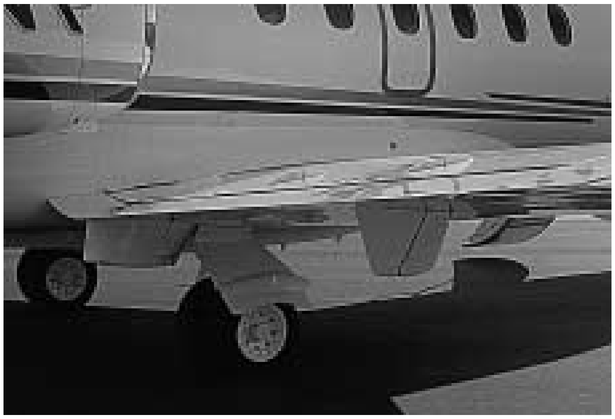
34. **UNDERSURFACE**—NO FUEL LEAKS, STALL VENT CLEAR



35. **VENT TANK INTAKE**—CLEAR

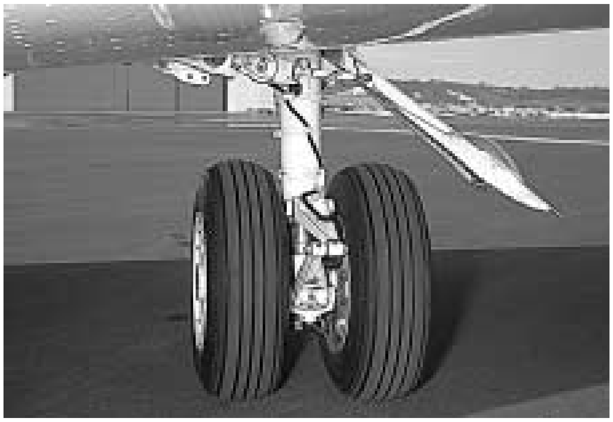


36. **AILERON**—CONDITION OF SURFACE
STATIC WICKS—SECURE AND UNDAMAGED



37. **FLAPS**-CONDITION OF SURFACE
38. **AIRBRAKES**-CONDITION OF SURFACE

RIGHT MAIN GEAR

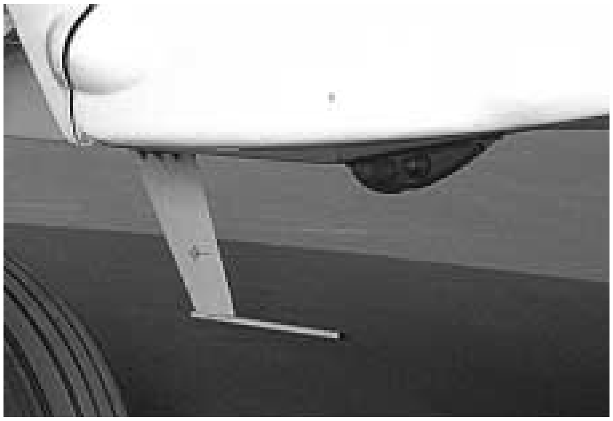


39. **FAIRING AND DOORS**—SECURE, UNDAMAGED
40. **WHEEL WELL**—GENERAL CONDITION, NO EVIDENCE OF FLUID LEAKS
41. **GROUND LOCK**—REMOVED
42. **STRUT**—CLEAN, NO FLUID LEAKS
43. **TIRES**—GENERAL CONDITION
44. **WHEELS**—UNDAMAGED
45. **BRAKE HOSES**—SECURE, UNDAMAGED, NO LEAKS

FUEL TANKS

46. **WATER DRAIN VALVES (5) (TYPICAL)**—OPERATE; ENSURE ALL WATER REMOVED

KEEL

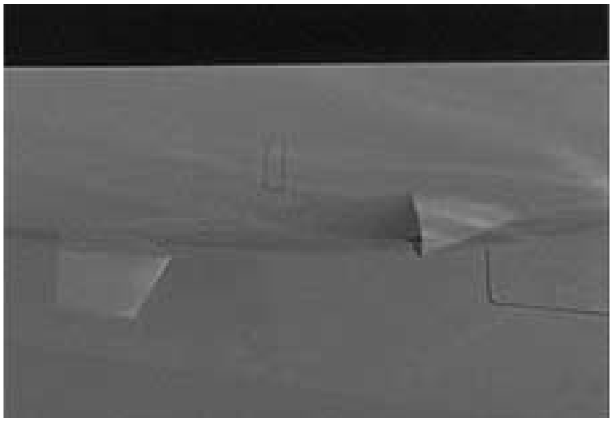


47. **ANTICOLLISION LIGHT**—SECURE, UNDAMAGED
48. **ANTENNAS**—SECURE, UNDAMAGED

RIGHT ENGINE

49. **COWLING AND LATCHES**—UNDAMAGED, FASTENERS SECURE, VENTS CLEAR, NO LEAKS

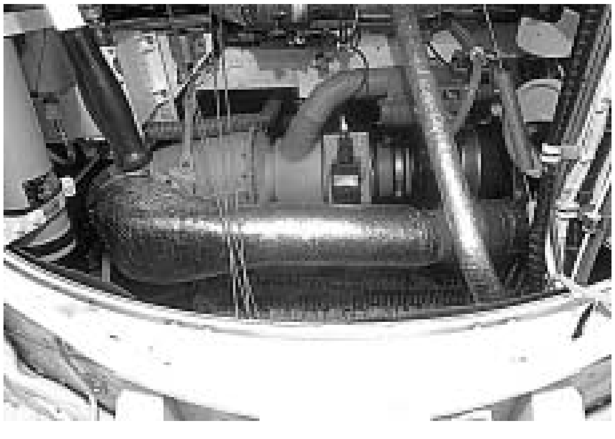
50. **TURBINE EXHAUST**—COVER REMOVED, CLEAR, DRY, UNDAMAGED
51. **THRUST REVERSER**—STOWED AND SECURE, PIN REMOVED



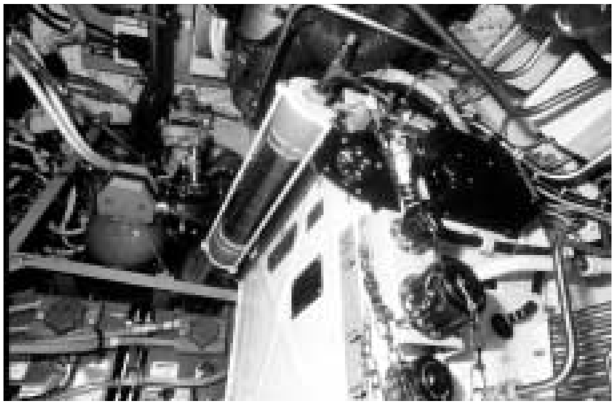
52. **OIL LEVEL**—CHECKED
53. **OIL FILTER**—CHECKED
54. **DRAIN MAST**—CLEAR, NO LEAKS

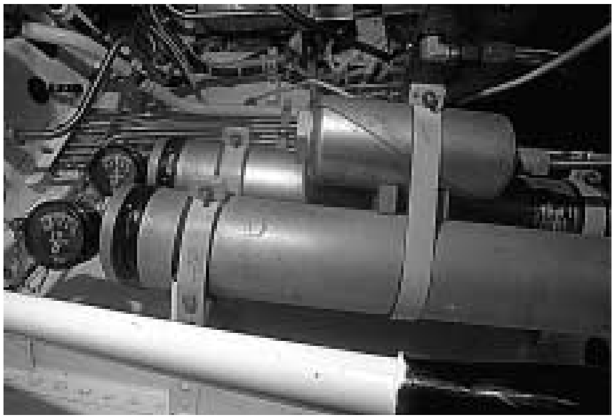
REAR EQUIPMENT BAY

55. **DOOR**—OPEN, UNDAMAGED



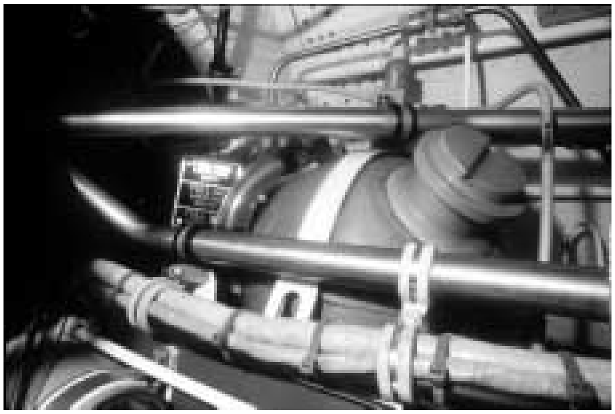
56. **AIR DUCTS**—GENERAL CONDITION





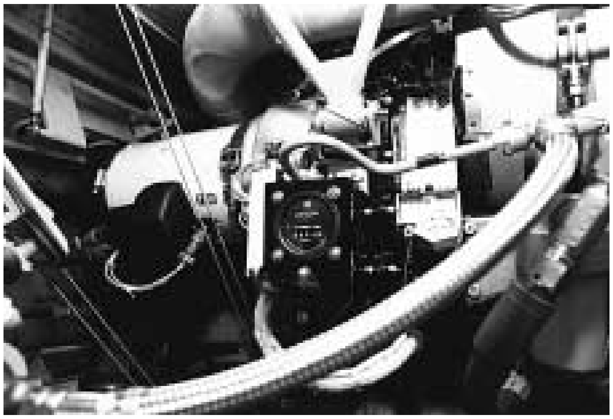
57. **HYDRAULIC ACCUMULATOR**—CHECK:

TR	1,450 PSI MINIMUM
MAIN	950 PSI MINIMUM
EMERGY	950 PSI MINIMUM



58. **HYDRAULIC RESERVOIR**—CONTENTS APPLICABLE TO ACCUMULATOR PRESSURE, NO LEAKS;
HYDRAULIC TANK PRESSURE GAGE—CHECK PRESSURE;
LH, RH ENGINE CHIP DETECTOR-BLACK
HYDRAULIC PRESSURE FILTER BYPASS BUTTON—IN

59. BATTERIES—CONNECTED AND SECURE
60. COMPUTERS—CONNECTED AND SECURE
61. ACCESS PANELS—SECURE

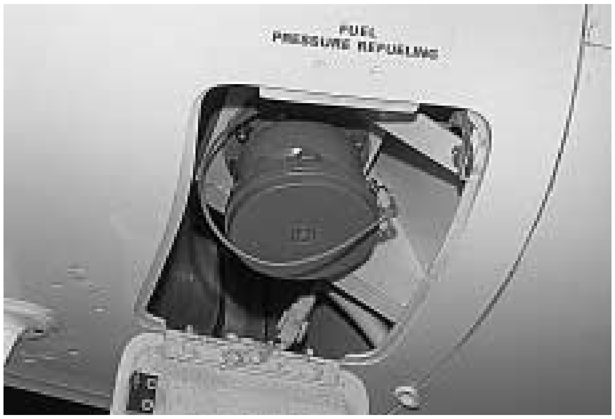


63. APU—GENERAL CONDITION, NO LEAKS
64. ALL LOOSE EQUIPMENT—REMOVED

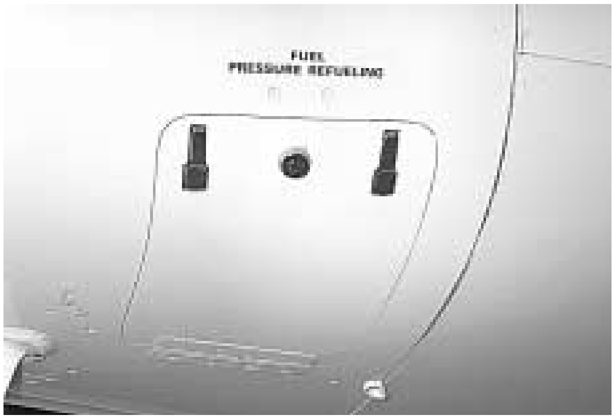
62. STICK PUSHER ASSEMBLY—GENERAL CONDITION, NO LEAKS

65. DOOR—CLOSED AND LOCKED

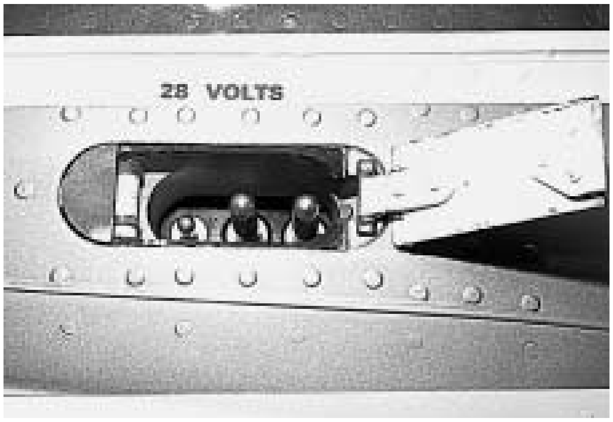
REAR FUSELAGE



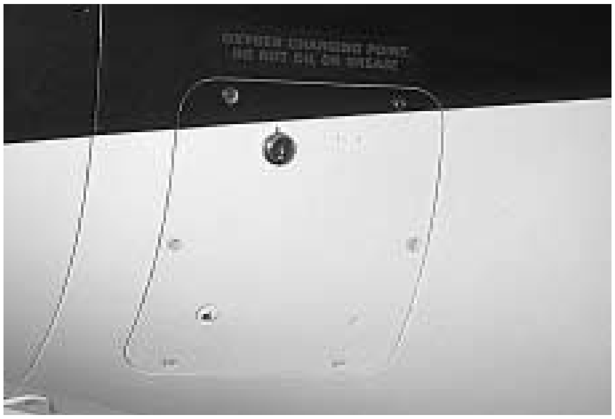
66. PRESSURE REFUELING RECEPTACLE (DOOR OPEN)—CAP INSTALLED, GENERAL CONDITION



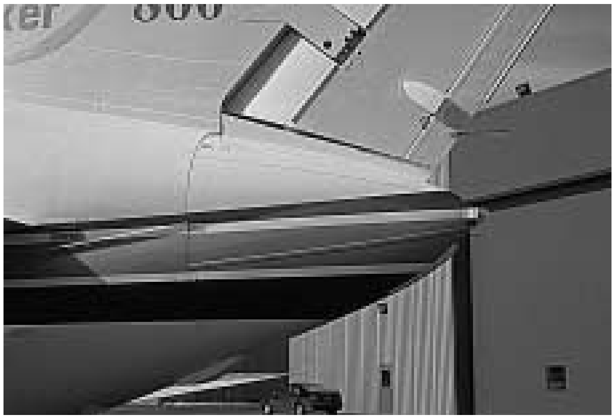
67. PRESSURE REFUELING DOOR—CLOSED, SECURE



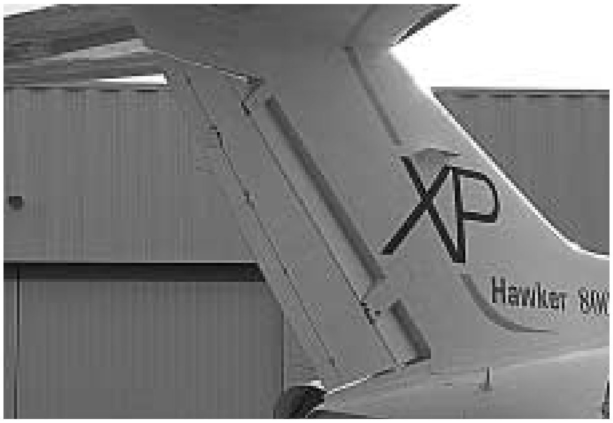
68. GROUND POWER—PROPERLY CONNECTED (IF REQUIRED) OR DOOR SECURE



69. OXYGEN CHARGING PORT—ACCESS PANEL SECURE



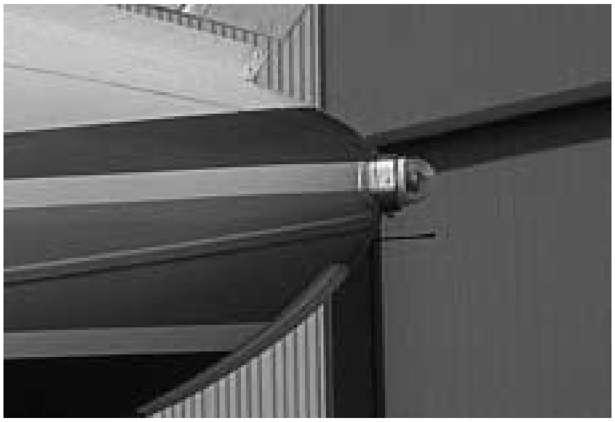
70. TAIL CONE—LOCKED, SECURE



71. VERTICAL STABILIZER AND RUDDER—
CONDITION OF SURFACES AND TABS
STATIC WICKS—PRESENT AND UNDAMAGED

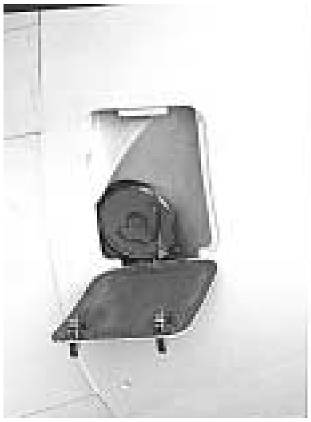
72. HORIZONTAL STABILIZER AND ELEVATORS—
CONDITION OF SURFACES

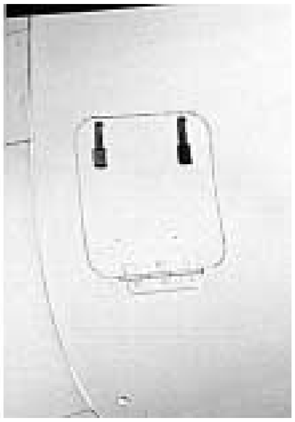
73. ANTICOLLISION LIGHT (TOP)—CHECKED
STATIC WICKS—PRESENT AND UNDAMAGED



74. TAIL CONE—STROBE AND BEACON LIGHTS, CHECKED

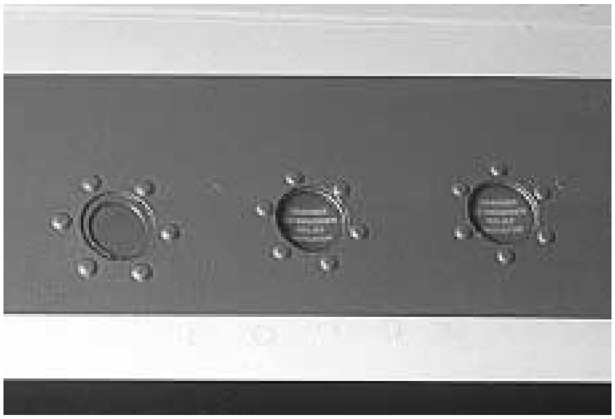
75. TAIL CONE VENTS—UNOBSTRUCTED





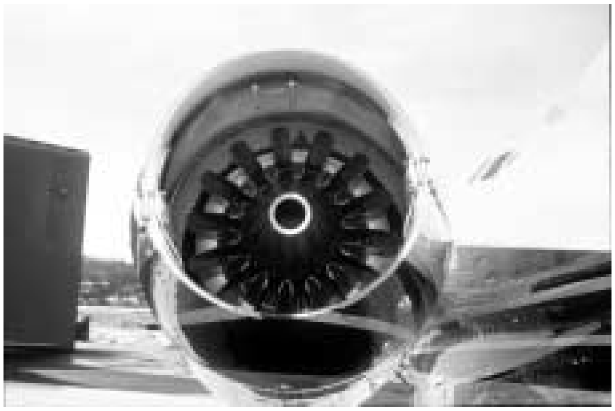
76. VENTRAL TANK—CAP SECURE, LOCKING LEVER IN PLACE

77. VENTRAL TANK ACCESS DOOR—CLOSED



78. FIRE EXTINGUISHER INDICATORS—INSTALLED (3)

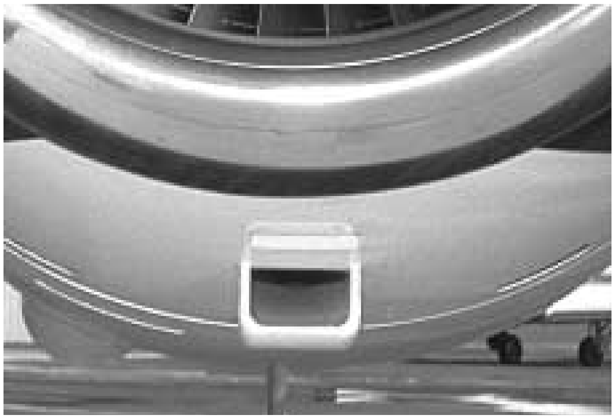
LEFT ENGINE



79. TURBINE EXHAUST—COVER REMOVED, CLEAR, DRY, UNDAMAGED

80. THRUST REVERSER—STOWED AND SECURE, PIN REMOVED

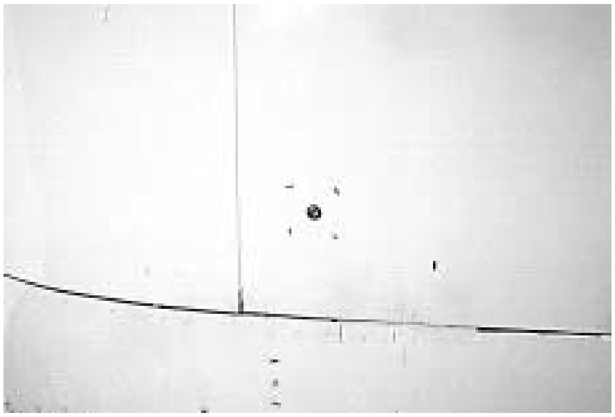
81. **OIL LEVEL**—CHECKED
82. **OIL FILTER**—CHECKED
83. **DRAIN MAST**—CLEAR, NO LEAKS



84. **STARTER**—GENERATOR COOLING AIR INTAKE-UNOBSTRUCTED

85. **FAN BLADES**—UNDAMAGED
86. **P_2T_2 SENSOR**—CLEAN, UNDAMAGED, SECURE
87. **INTAKE**—COVER REMOVED, NO FOREIGN OBJECT DAMAGE

FUEL TANKS

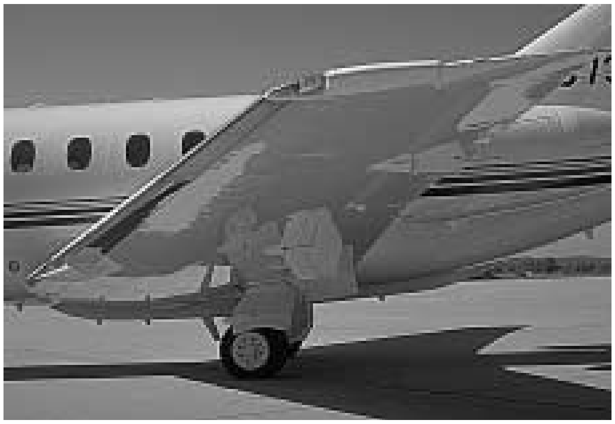


88. **WATER DRAIN VALVES (5) (TYPICAL)**—OPERATE; ENSURE ALL WATER REMOVED

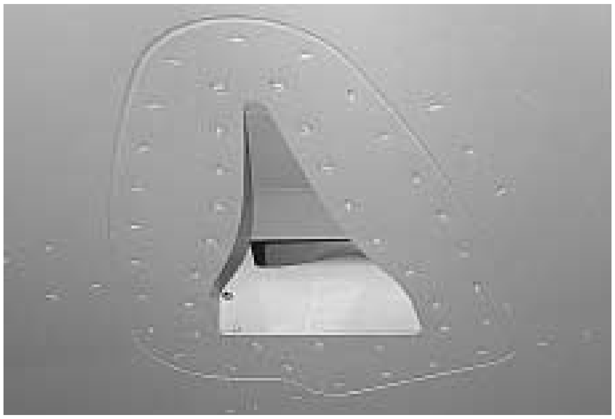
LEFT MAIN GEAR

89. **FAIRING AND DOORS**—SECURE, UNDAMAGED
90. **WHEEL WELL**—GENERAL CONDITION, NO EVIDENCE OF FLUID LEAKS
91. **GROUND LOCK**—REMOVED
92. **STRUT**—CLEAN, NO FLUID LEAKS
93. **TIRES**—GENERAL CONDITION
94. **WHEELS**—UNDAMAGED
95. **BRAKE HOSES**—SECURE, UNDAMAGED, NO LEAKS

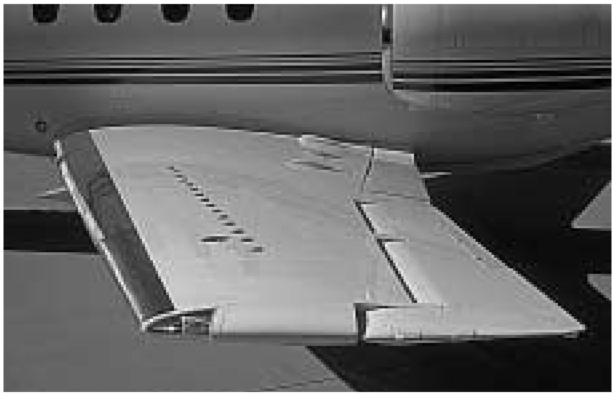
LEFT WING



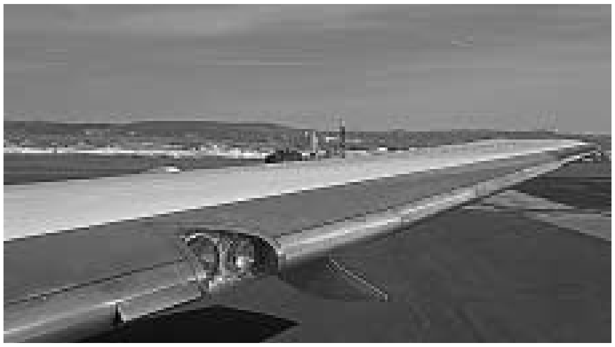
96. **UNDERSURFACE**—NO FUEL LEAKS, STALL VENT CLEAR



97. **VENT TANK INTAKE**—CLEAR

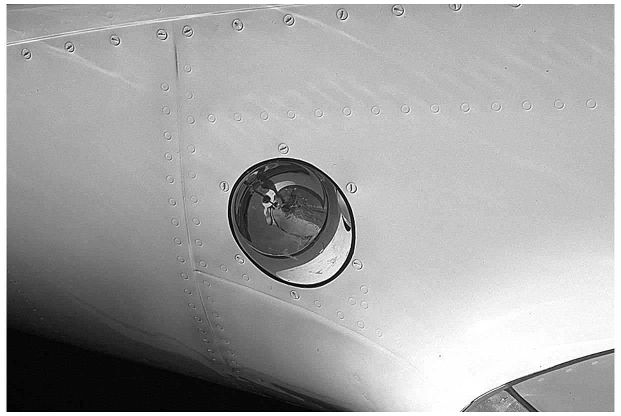


98. **AIRBRAKES**—CONDITION OF SURFACE
99. **FLAPS**—CONDITION OF SURFACE
100. **AILERON**—CONDITION OF SURFACE
101. **VORTEX GENERATORS**—SECURE
102. **FUEL CAPS**—SECURE
103. **NAVIGATION AND STROBE LIGHTS**—COVER CLEAN, UNDAMAGED

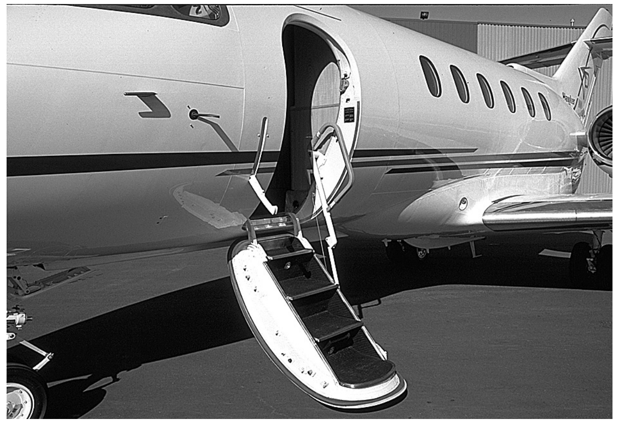


104. **VORTILON**—SECURE, UNDAMAGED
105. **LANDING AND TAXI LIGHTS**—CONDITION OF LIGHTS, COVER CLEAN, UNDAMAGED
106. **STALL STRIP**—SECURE, UNDAMAGED
107. **TKS PANELS**—CLEAN, UNDAMAGED
108. **TOP SURFACE**—GENERAL CONDITION

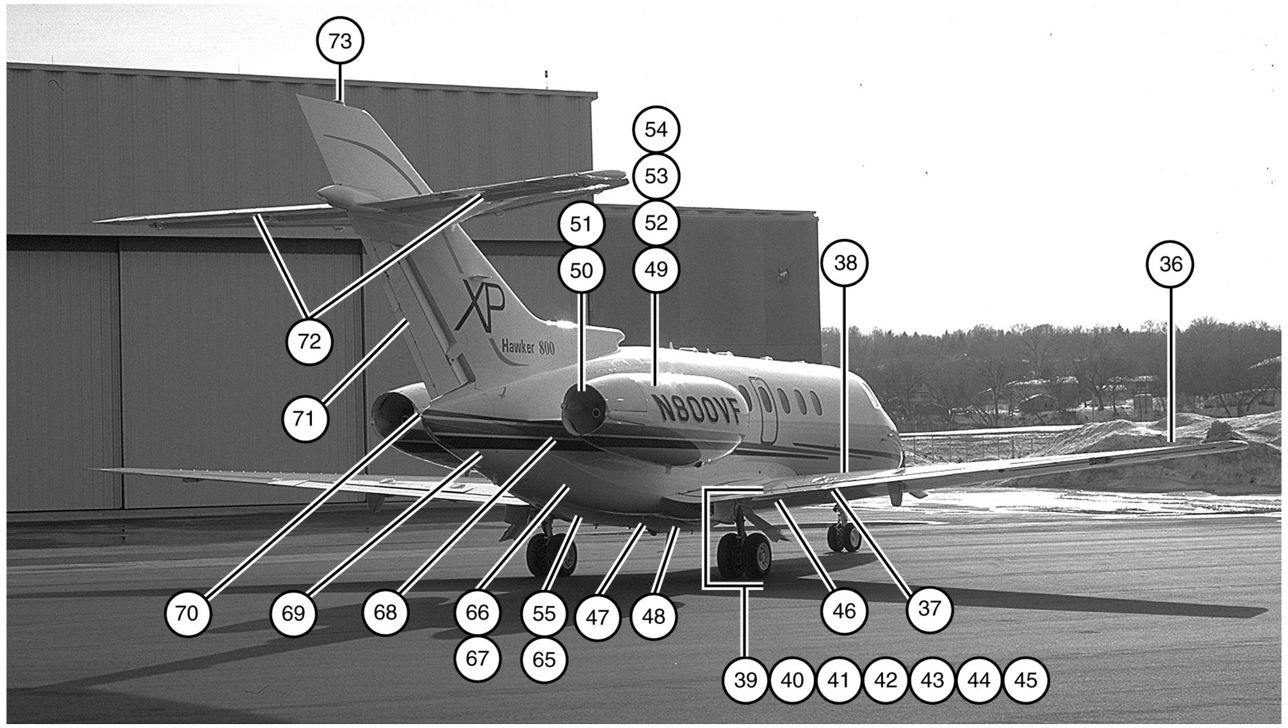
LEFT FUSELAGE

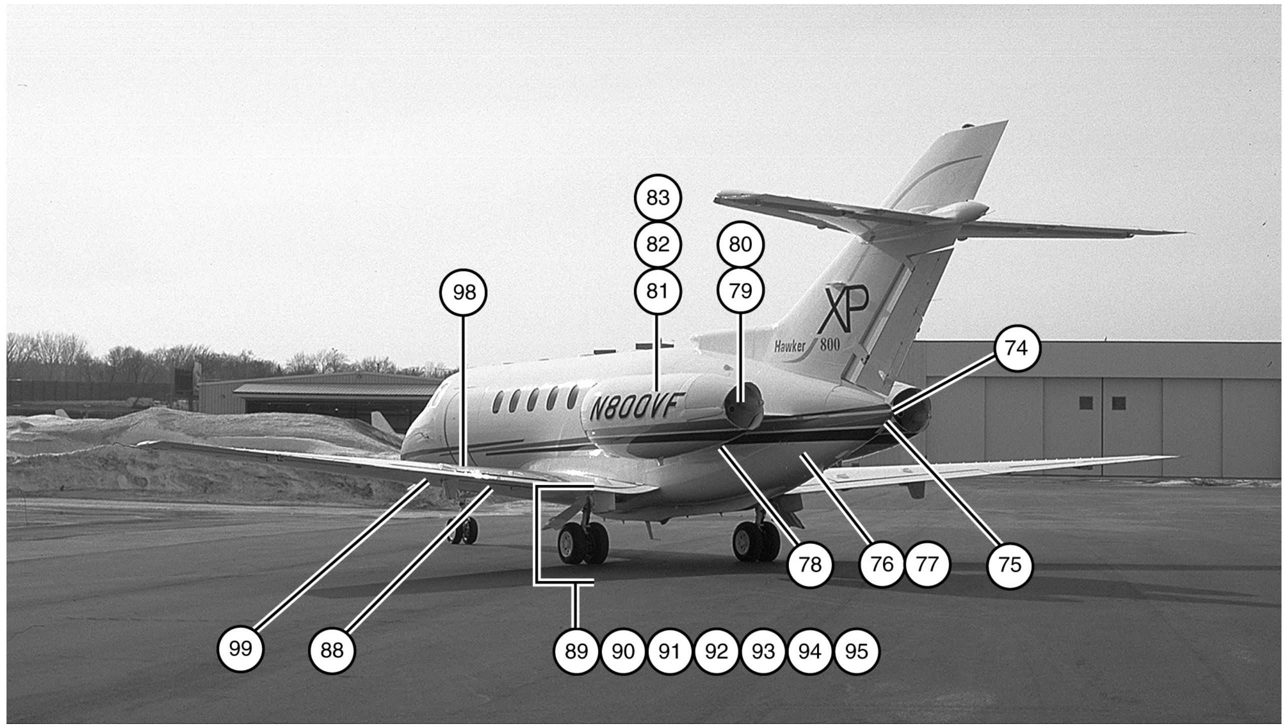


109. WING ICE INSPECTION LIGHT—CHECKED



110. ENTRANCE DOOR—SEAL SECURE; HANDRAIL LINKAGE SECURE, UNDAMAGED; GENERAL CONDITION





ANNUNCIATORS

The following section presents a color representation of all the annunciator lights in the airplane.

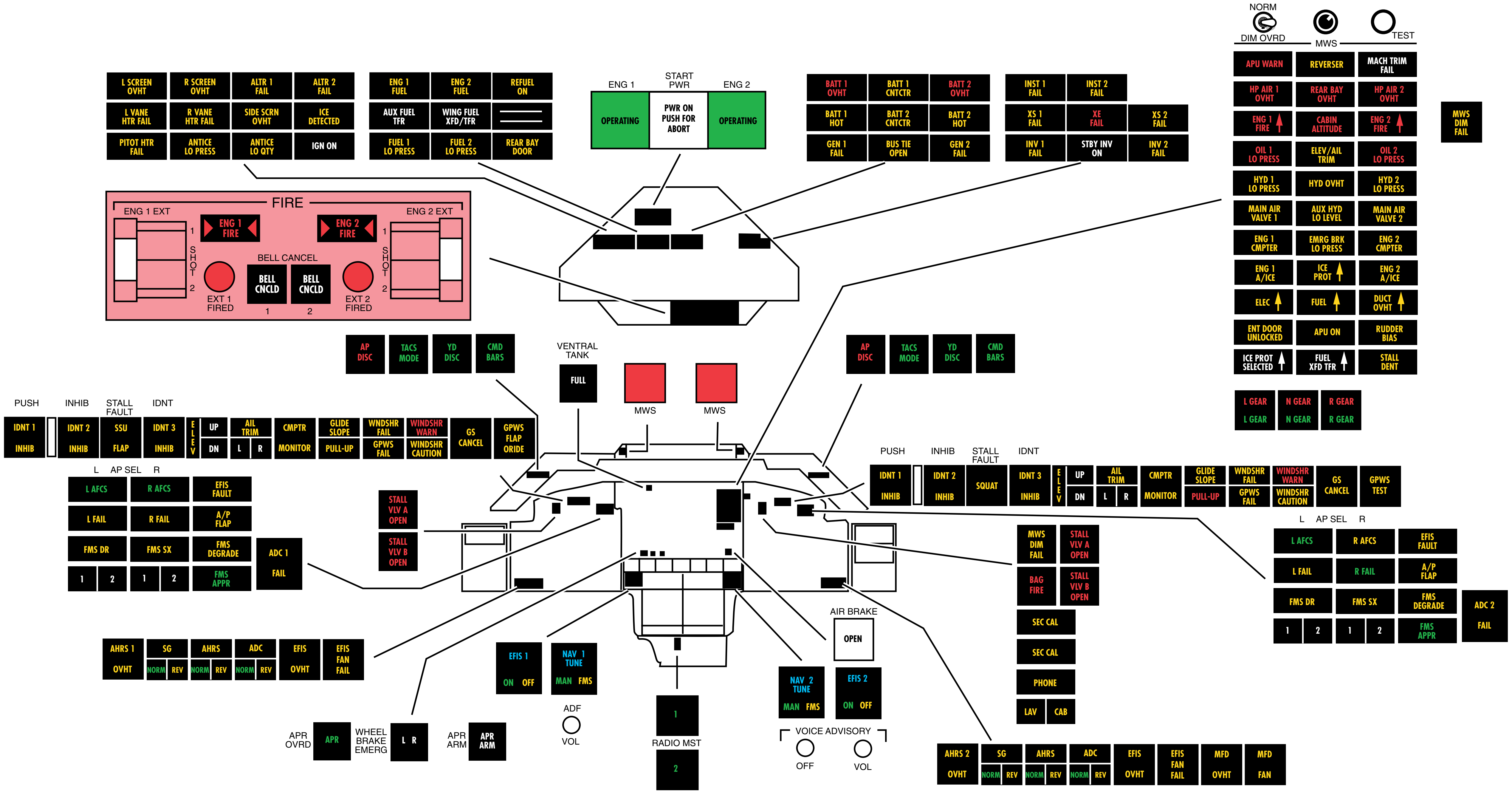


Figure ANN-1. Annunciators